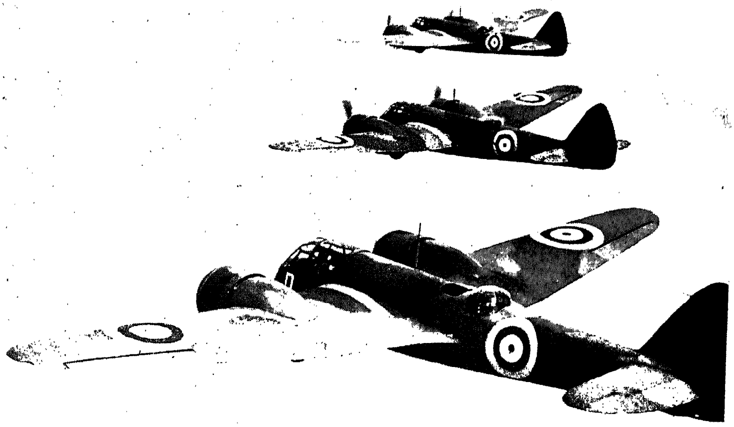

BRISTOL "BLENHEIMS" IN FLIGHT
TWO "MERCURY XIII" ENGINES
("*Flight*" *Photo*)

[*Frontispiece*

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AIRCRAFT DESIGN

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VOLUME ONE

AERODYNAMICS

LONDON

CHAPMAN & HALL LTD.

11 HENRIETTA STREET, W.C.2.

1939

First Published, 1939
by CHAPMAN & HALL, LTD., 11 *Henrietta Street*, LONDON, W.C. 2
Printed in Great Britain at the BURLEIGH PRESS, *Lewin's Mead*, BRISTOL
Bound by C. & J. KITCAT LTD., LONDON
Flexiback Binding Patent No. 441294

PREFACE

THIS book has been written to meet the need for an up-to-date work dealing with the general principles of aircraft design, and is intended to be of use both as a textbook for serious students of the subject and as a guide for the increasing number of private constructors of aeroplanes. It is based on experience in the design, construction, and piloting of aircraft, and lecturing in aeronautical subjects, during a period of over twenty years.

In order that the process of design shall not be continually interrupted with explanations of the principles underlying the various features of design and the function of the many devices deemed necessary for incorporation in present-day aircraft, the book has been divided into two volumes; the first of which outlines in simple language the principles of flight and stability, whilst the second deals mainly with the mathematical treatment of design.

Volume I includes a chapter of some length devoted to an analysis of parasite drag: Materials, seaplanes, and experimental testing are dealt with in the second volume.

The author wishes to acknowledge his indebtedness to Mr. W. G. Sheppard, M.A., and to Mr. D. L. Rundle for valuable assistance in the preparation of this work, for the reading of proofs and for many helpful suggestions: also to the several firms who have kindly supplied photographs, and to *Flight*, *The Aeroplane* and *Practical Mechanics* for the use of diagrams and blocks. Free use has been made of existing aeronautical books and in particular the *Journal* of the Royal Aeronautical Society; Warner's "Aeroplane Design"; Air Publications 970 and 1208; Reports and Memoranda of the British Aeronautical Research Committee, by kind permission of the Controller of H.M. Stationery Office, and the various reports of the American National Advisory Committee for Aeronautics; for all of which the author expresses his gratitude.

Uxbridge,
December, 1938.

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LIST OF SYMBOLS USED IN THIS VOLUME

P = force, lb.
M = moment of a force, lb. ft.
R = resistance of a body in a moving fluid (relative) lb. *also*
= reaction of aerofoil in a moving fluid (relative) lb.
L = lift force, lb.
L_P = profile lift of aerofoil.
D = drag force of aerofoil, lb.
D_P = profile drag of aerofoil.
D' = parasite drag of aircraft, lb.
D_t = total drag $= D + D'$, lb.
D_i = induced drag of aerofoil, lb.
D_f = skin friction drag.
W = weight of aircraft, lb.
a = acceleration, ft. per sec.^{2}
g = acceleration due to gravity = 32·2 ft. per sec.^{2}
$\rho = \dfrac{\text{weight of cu. ft. air}}{g}$ = fluid density, slugs per ft.^{3}
w = unit weight, i.e., per sq. ft. or per cu. ft., lb.
p = pressure, lb. per sq. ft.
v = volume, ft.^{3}
t = temperature, degrees.
V = velocity, ft. per sec. (or m.p.h. where stated).
V_s = stalling velocity, ft. per sec.
V_c = rate of climb, ft. per sec.
c = chord of aerofoil, ft.
b = span of aerofoil, ft.
S = cross sectional area of moving object, ft.^{2} *also*
= area of aerofoil, ft.^{2}
A = aspect ratio of aerofoil $= \dfrac{b^2}{S}$.
α = angle of incidence of aerofoil, degrees.
ϵ = angle of downwash, degrees or radians.
γ = angle of glide, degrees.
C_P = centre of pressure.
C_G = centre of gravity.
C.P.F. = centre of pressure forward.
C.P.B. = centre of pressure back.
L.N.D. = limiting nose dive (= T.V.D. = terminal velocity dive).
C_M = moment coefficient.
C_R = coefficient of resistance.
C_L = coefficient of lift.
C_D = coefficient of drag.
$C_{D'}$ = coefficient of parasite drag.

LIST OF SYMBOLS USED IN THIS VOLUME

C_{Dt} = coefficient of total drag ($= C_D + C_{D'}$).
C_{Di} = coefficient of induced drag.
C_{Df} = skin friction drag coefficient.
μ = coefficient of fluid viscosity.
ν = coefficient of kinetic viscosity $= \frac{\mu}{\rho}$.
C_{CP} = centre of pressure coefficient.
R.N. = Reynolds' number $= \frac{Vl}{\nu}$.

AIRSCREW

P = brake horse power.
T = thrust, lb.
Q = torque, lb. ft.
D = diameter, ft.
p_g = geometric pitch, ft.
p_e = experimental pitch, ft.
n = revolutions per second.
V_s = slipstream velocity, ft. per sec.
C_T = thrust coefficient.
C_Q = torque coefficient.
η = efficiency.

R.A.F. = Royal Aircraft Factory, Farnborough (*Now* Royal Aircraft Establishment, R.A.E.).
N.P.L. = National Physical Laboratory, Teddington.
N.A.C.A. = National Advisory Committee for Aeronautics (U.S.A.).
R. & M. = Reports and Memoranda published by the Aeronautical Research Committee.

Note.—British reports prior to March, 1936, made use of aerodynamic coefficients symbolised by the letter K and having half the value of C coefficients adopted at that time.

$$\left(\text{e.g., } C_L = \frac{L}{\frac{\rho}{2} SV^2} \text{ in place of } K_L = \frac{L}{\rho SV^2}.\right)$$

When such reports are referred to it is an easy matter to double all K values and substitute the symbol C. American and Continental reports are based on the stagnation pressure, $\frac{\rho}{2}V^2$, as now adopted in this country, and such reports may therefore be accepted without alteration.

VOLUME I

CHAPTER I

GENERAL PRINCIPLES

The Atmosphere.

The atmosphere is described as the whole mass of aeriform fluid surrounding the earth. It is a gaseous matter attracted by the force of the earth's gravitation, and thus takes the shape of a thick, hollow sphere with the earth as core.

Most means of locomotion take place, in part at least, through the air, and are therefore affected to some extent by its properties, but in the case of aircraft the vehicle is not only entirely surrounded by air, but also obtains it's support, and generally thrust, from the air.

Furthermore, aerial support and propulsion are closely associated with high speeds of movement, being in fact, with the exception of the sustentation of lighter-than-air craft, entirely dependent on this factor, and for this reason a careful study of the properties and behaviour of air is of great importance to the aeronautical engineer.

Properties of Air.

Like all gases, air is compressible and the part nearest to the earth's centre, that is the air at ground, or sea, level, having the weight of the upper air to support, becomes compressed and has therefore greater density and weight.

An important point to note here is that flight generally, at speeds below the velocity of sound (1,118 ft. per sec. at sea level, and normal temperature), does not cause any substantial compression of the air, or to be more exact, the effects of compressibility are of little consequence in the consideration of the air force's acting on a wing, although when dealing with airscrews, having tip speeds in the neighbourhood of the speed of sound,

the effects of air compression assume an importance that cannot be neglected.*

The pressure of air at ground level is 14·7 lb. per square inch, which is another way of saying that a vertical column of air, of one square inch cross-sectional area, weighs 14·7 lb. It is well-known that a fluid exerts the same pressure in all directions, at a constant depth, and, in obedience to this law, the pressure exerted on, say, the wall of a building is the same as that exerted on the floor, or ceiling (neglecting any slight difference in height). This will be found of importance later when the forces acting on an aeroplane wing are considered.

THE WEIGHT OF AIR.—At sea-level the weight of air is 0·0000439 lb. per cubic inch, or 0·077 lb. per cubic ft., the latter being the form mostly used in aerodynamics.

The effects of alteration in density and temperature with height are of importance when considering the performance of aircraft and engines at varying heights above the earth and, in reality the same thing, their performance when operating from aerodromes at considerable heights above sea level. And, furthermore, it is desirable that all performances should be relative to some common basis for purposes of comparison, and for this reason figures for an *International Standard Atmosphere* have been agreed upon. The variation of density and temperature with height are shown in Table I, from which it will be noticed that temperature decreases with height at a steady rate of 20° C. for every 10,000 ft.

At about 36,000 ft., however, the temperature ceases to fall off and remains constant thereafter at − 56·5° C. All above this height is called the *stratosphere,* and below is termed the *troposphere.*

Actual conditions do not, however, comply with these rigid gradings, there being fluctuations in different parts of the world and at different seasons, whilst the little data at present available relative to the stratosphere show both decreases and increases of temperature, chiefly the former, from the hypothethical standard of − 56·5° C.

VISCOSITY.—A further property of the air of importance to flight is viscosity, which may be defined as the tendency for particles of a fluid to cling together. Air-flow, as represented

* See p. 166.

by wind, or the flow past an aeroplane in flight (relatively), tends to slow down in the neighbourhood of any object, or surface, according to the degree of roughness of the surface. Thus the wind velocity near the ground is slowed down, and similarly the air adjacent to the surfaces of an aeroplane gets dragged along with the machine to some extent.

TABLE I.—I.C.A.N. STANDARD ATMOSPHERE

Height ft.	*Temp. deg. C.*	*Relative Pressure*	*Relative Density*	*Height ft.*	*Temp. deg. C.*	*Relative Pressure*	*Relative Density*
0	15	1	1	31,000	−46·5	0·284	0·360
1,000	13	0·964	0·971	32,000	−48·5	0·271	0·347
2,000	11	0·930	0·943	33,000	−50·5	0·258	0·334
3,000	9	0·896	0·915	34,000	−52·5	0·246	0·322
4,000	7	0·864	0·888	35,000	−54·5	0·235	0·310
5,000	5	0·832	0·862	36,000	−56·5	0·224	0·295
6,000	3	0·801	0·836	37,000	−56·5	0·213	0·281
7,000	1	0·772	0·811	38,000	−56·5	0·204	0·268
8,000	− 1	0·743	0·786	39,000	−56·5	0·195	0·255
9,000	− 3	0·715	0·762	40,000	−56·5	0·185	0·245
10,000	− 5	0·688	0·738	41,000	−56·5	0·176	0·232
11,000	− 7	0·661	0·715	42,000	−56·5	0·168	0·221
12,000	− 9	0·636	0·693	43,000	−56·5	0·160	0·211
13,000	−11	0·611	0·671	44,000	−56·5	0·153	0·202
14,000	−12·5	0·587	0·650	45,000	−56·5	0·146	0·192
15,000	−14·5	0·564	0·629	46,000	−56·5	0·140	0·184
16,000	−16·5	0·542	0·609	47,000	−56·5	0·133	0·175
17,000	−18·5	0·520	0·589	48,000	−56·5	0·126	0·167
18,000	−20·5	0·499	0·570	49,000	−56·5	0·120	0·160
19,000	−22·5	0·479	0·551	50,000	−56·5	0·115	0·152
20,000	−24·5	0·459	0·533	51,000	−56·5	0·110	0·145
21,000	−26·5	0·440	0·515	52,000	−56·5	0·105	0·138
22,000	−28·5	0·422	0·498	53,000	−56·5	0·100	0·132
23,000	−30·5	0·404	0·481	54,000	−56·5	0·095	0·126
24,000	−32·5	0·387	0·464	55,000	−56·5	0·090	0·120
25,000	−34·5	0·371	0·448	56,000	−56·5	0·086	0·114
26,000	−36·5	0·355	0·432	57,000	−56·5	0·082	0·108
27,000	−38·5	0·340	0·417	58,000	−56·5	0·078	0·104
28,000	−40·5	0·325	0·402	59,000	−56·5	0·075	0·097
29,000	−42·5	0·311	0·387	60,000	−56·5	0·0715	0·094
30,000	−44·5	0·297	0·374				

For convenience, air-flow is looked upon as taking the form of layers, or laminæ, and it will be at once seen that the slowing up of one layer must retard the layer next to it, but to a lesser extent, and so on.

WIND.—When a large volume of air moves over the earth's surface, it is referred to as wind, and since the air constitutes the supporting medium of all aircraft, it is obvious that movement of the air must be accompanied by a similar movement (relative

to the earth) of everything depending for support on it, e.g., birds, balloons, and aeroplanes.

This horizontal movement of the air is exploited by man and bird alike, but it also acts heavily against them. Assistance from the wind is generally obtained during the take-off and alighting, most long distance flights have been made *with* the wind, whilst without the use, during ascent or descent, of special directing surfaces, free balloon flights across country would be impossible in the absence of wind. But, against this, the progress of powered aircraft *against* the wind is retarded by the wind's movement, and unfortunately the loss on the roundabouts is greater than the gain on the swings.

A simple calculation shows that an out and return flight, made in *any* conditions of wind, must result in a loss of time in comparison to the same journey made in the absence of wind. Suppose, for instance, two points, A and B, are 100 miles apart and a wind of 20 m.p.h. is blowing from B to A. An aeroplane flying at 50 m.p.h. would take $\frac{100}{50 - 20} = 3\frac{1}{3}$ hr. to fly from A to B, and $\frac{100}{50 + 20} = 1\frac{3}{7}$hr. on the return journey. Thus the total time is $4\frac{16}{21}$ hr. against 2 hr. each way, when no wind is present. Similarly a side wind increases the duration of a flight, and again a slow-moving aircraft in a wind of greater velocity than that of the aircraft would be unable to make a return journey at all.

Air movements are not limited to horizontal directions but generally have vertical components as well, and this also has been responsible for success and disaster alike. Practically all soaring flight is carried out by making use of the vertically upward components of wind, whilst its aid is often sought by pilots of heavily laden aircraft in crossing over mountains and other obstacles. Down currents, on the other hand, spell disaster and should be avoided.

Another important feature of up and down currents of air is their effect on the structural strength of aircraft, for an aeroplane changing suddenly from a region of horizontally moving air to one in which the air has a vertical component is subjected to very changed conditions of loading, which have been sufficient in some cases to cause structural failure. This is dealt with in a later chapter.*

* Chap XI, p. 153. See also Vol. II, Chap. III.

Action and Reaction.

This is the fundamental principle of heavier-than-air flight: In order that an aeroplane may remain at a constant height, the downward attraction of gravity must be counteracted by means of a supporting force. This, of course, is Newton's Second Law.

The supporting force can only be provided by giving a downward acceleration to some other body, or mass, which in this case is the air. It has been seen that the density of air is relatively low; that is, its mass per unit volume is small, and therefore it is obvious that either:

(*a*) A relatively small quantity of air must be given a high downward acceleration, or
(*b*) A large quantity of air must be acted upon.

The second is the method employed in present-day flight, although the mass and acceleration vary within certain limits throughout the range of flight speeds. In order that a large mass of air may be acted upon in unit time the span and speed of the aerofoil must both be high. The slowest speed of flight (stalling speed) for any given span, is reached when the air is being deflected with the maximum downward acceleration and the mass acted upon per unit of time is then a minimum.

To support the aeroplane in horizontal flight, the upward force P must be equal to the weight of the aeroplane, W, and to the downward force imparted to the air, or $P = W = \frac{W_1}{g} a_1$ where W_1 and a_1 are the weight of air acted upon and the average acceleration induced.

The mass of air acted upon is then $\frac{W_1}{g} = \frac{W}{a_1}$

Or, in words, the weight of air forced downwards is equal to the weight of the aircraft multiplied by g/a_1. If the weight of air, W_1, is decreased, the acceleration, a_1, must increase and *vice versa*.

The following simplified example will help to make this clear. Suppose an aeroplane is moving horizontally at a speed of 100 ft. per sec. (see Fig. 1), and its span and weight are 40 ft. and 1,000 lb. respectively.

It has been found that the volume of air acted upon is roughly equal to a horizontal cylinder of diameter equal to the span.

Therefore area of air cylinder $= \times (40)^2 = 1{,}256$ sq. ft., say.

Volume of air acted upon $= 1{,}256 \times 100 = 125{,}600$ cu. ft. per sec. and weight $= 125{,}600 \times \cdot 077 = 9{,}670$ lb. per sec.

Average downward acceleration $= \dfrac{1{,}000 \times 32{\cdot}2}{9670} = 3{\cdot}34$ ft. per sec.^{2}

As a rough explanation only, sufficient for the present stage, the passage of the wing through the air may be regarded as deflecting the air within its vicinity to a lower level, as has been indicated in the diagram.

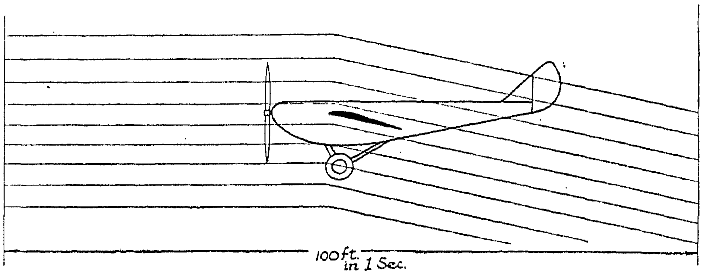


FIG. 1

Effect of Temperature.

The temperature of the air decreases, generally, with height. This is because the air's heat is obtained chiefly through convection from the earth's surface, where the heat emitted by the sun is stored up, only very little being directly extracted from the sun's rays by radiation. The effect of this heating up is to cause the air to expand, and thus to reduce its density to some extent, particularly in the lower layers.

This action causes vertical movements of the air which extend to heights of several thousand feet, whilst colder, and therefore heavier, surrounding air moves in to take its place. Both vertical and horizontal air currents are thus set up.

Temperature, pressure and density are connected by Charles' Law for gases, which states that $\frac{pv}{t}$ remains constant, where p = pressure, v = volume, and t = temperature. Thus, if the values for these are known at sea level, and pressure and temperature are measured at some other height, the density at that height can also be found.

Fluid Resistance.

If a smooth-shaped body is moved through a "perfect" fluid, i.e., a fluid incompressible and having no viscosity, it can be shown that it would experience no resistance to motion. For the particles of the fluid in the path of the body would be moved away from their original positions during the body's passage and would return again afterwards. Again, there would be no translational motion of the particles in the direction of movement of the body, since this would entail friction, whilst there would likewise be no generation of heat, so that the net work done must be nil.

In the case of an imperfect fluid, such as air, friction exists between the body and fluid, and between the fluid particles themselves, with the result that the fluid coming within the influences of the disturbance is given a forward momentum, whilst rotation, or eddy motion, may be caused, due to viscosity.

The resistance met with by the body depends on the properties of the medium in which it moves, and obviously on its own size, surface texture, and shape.

The cylinder of fluid influenced by the passage of the body may be regarded as a jet moving against the body, and the force exerted must be equal and opposite to the reaction on the fluid. If it is assumed that the whole of the fluid cylinder is brought to rest as it meets the body, the force exerted thereby is equal to the kinetic energy of the moving mass given up per unit of time, i.e., $\frac{1}{2}\ \rho\ S\ V^2$, where S is the cross-sectional area of the obstruction causing stagnation of the fluid cylinder, ρ is the fluid density, and V is the initial velocity, and the pressure is $\frac{1}{2}\ \rho\ V^2$ per unit of area.

Alternatively, if the fluid is merely assumed to be turned aside, through an angle of 90 degrees, with no change of speed, then the momentum in the original direction, $\rho\ S\ V^2$, is destroyed and the pressure is given by $\rho\ V^2$, or twice the previous value.

In practice, a small area gives a result agreeing very closely with the kinetic pressure, and this result is made use of in the case of the pitot, or open-ended pressure tube, used for measuring air speed, but as the surface (perpendicular to the air-stream) becomes larger the pressure tends to increase to the double value.

With bodies of streamline and other similar shapes, e.g., aerofoils, fuselages, nacelles, the fluid is pierced by the passage

of the body, moves outward relative to the latter's axis of motion, and closes in again afterwards with variation in both velocity and pressure, but there is one point on the surface, known as the stagnation point, at which the fluid is given a velocity in the direction, and equal to that, of the body's motion, and at which the pressure is accurately given by $\frac{1}{2} \rho V^2$.

Scale Effect and Reynolds' Number.

It has been found that, for ordinary speeds of flight at least, the force of resistance of a fluid to the passage of a body is proportional to $\frac{1}{2} \rho S V^2$,

$$\text{or} \quad R = C_R \frac{\rho}{2} S V^2 \quad . \quad . \quad . \quad . \quad . \quad . \quad (1)$$

where C_R is a coefficient of resistance depending on the form of the body, and of course on the flow pattern accompanying such forms, and is found for each shape by experiment. But the flow pattern does not remain constant at all speeds and therefore the value of C_R changes also. For instance, at very low speeds the flow conforms itself without difficulty to the outline of the body, but at higher speeds the momentum imparted to the fluid is greater and eddy formation results (this can be clearly observed in the case of slow-moving and rapid water streams and rivers), whilst again, at very high speed, changes of density due to compression take place.

As already explained, except for isolated problems, the effect of compressibility need not be considered in connection with present-day flight. At the lower end of the scale, the smooth flow effect already mentioned relates to speeds below those used in ordinary flight, and it is therefore seen that over the range of flight speeds in common use to-day the resistance formula $R = C_R \frac{\rho}{2} S V^2$ is tolerably correct.

Owing chiefly to the large quantity of wind tunnel data on which the aeroplane designer depends, and the fact that so much wind tunnel work is carried out under conditions very different to those obtaining in ordinary full-scale flight, the validity of the simple resistance formula can no longer be relied upon.

A more accurate formula, but still neglecting the effect of compressibility, covering the wide range of conditions connecting small model work with normal flight, is :

$$R = \frac{\rho}{2} S V^2 \times f\left(\frac{\rho V l}{\mu}\right), \quad . \quad . \quad . \quad . \quad (2)$$

where l is some convenient linear dimension of the body, and μ is the coefficient of fluid viscosity.

It will be seen that the coefficient C_R of the earlier formula has been replaced by the function   It is usual to substitute ν for μ/ρ, ν being termed the coefficient of kinematic viscosity, so that the expression becomes $R = \frac{\rho}{2} S V^2 \times f\left(\frac{V l}{\nu}\right)$. (3)

The value $\frac{V l}{\nu}$ is called the *Reynolds' number,* and for conditions of true similarity in two tests, or in a model test and full-scale flight, the Reynolds' number should be constant. This being so the original simple form of the resistance equation may be reverted to.

When making use of wind tunnel data for application to full-size design, care should be taken to ensure that the data were obtained at the required Reynolds' number, or the aerodynamic values may be extrapolated from results for other available Reynolds' numbers, or again a scale correction may be applied.

Here it may be observed that each wind tunnel test is carried out at reasonably constant speed, with a constant value of Reynolds' number, but that in full-scale flight there is considerable variation in Reynolds' number between the slowest and top speeds of flight. This means that the values of the aerodynamic coefficient employed in calculations at the two ends of the speed range should be adjusted for Reynolds' number.

At normal sea-level temperature and pressure the value of ν is 0·000159, and as an indication of the values of Reynolds' numbers used in present-day flight it may be stated that a low figure of 1×10^6 represents a wing chord of about 3·5 ft. in flight at a velocity of 30 m.p.h.; a chord of, say, 7 ft., and the moderate speed of 150 m.p.h. gives a Reynolds' number of 10^7, whilst a 10-ft. chord moving at the high speed of 450 m.p.h. provides a Reynolds' number of roughly 4×10^7. It will be noticed that in the case of an aerofoil the chord is selected as the linear dimension, l.

The Wind Tunnel.

The wind tunnel consists essentially of a tubular structure through which a stream of air is forced (Fig. 2). Provision is made for the mounting of models inside the tunnel so that the effect on the model of the passage of the air-stream may be determined with a good degree of accuracy.

The air-stream is kept in motion by means of an airscrew, driven, generally, by means of an electric motor. In order to obtain increased speeds of flow the tunnel may take the shape of a venturi tube, the working section being positioned at the narrow part where velocity is greatest.

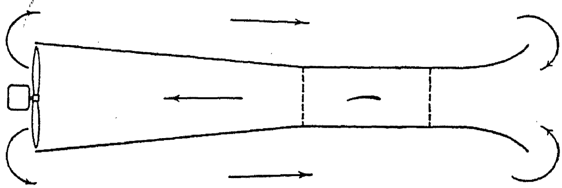


FIG. 2.—DIAGRAMMATIC SKETCH OF WIND TUNNEL

It will be appreciated that there must be a pressure drop over the restricted area, in compliance with the well-known characteristic of the venturi tube,* but this is relatively small and does not have any appreciable effect on most experiments, since there is uniform pressure surrounding the model. For experiments connected with pressure plotting over aerofoils, or other objects, the actual pressure in the working portion of the tunnel is recorded and used as datum in place of atmospheric pressure.

Open and Closed Jets.

Wind tunnels may be divided into two main groups. In the closed jet type the tunnel walls are continuous from end to end, as shown in Fig. 2. Access to the working part is gained by means of doors, whilst observation is made possible by glass panels. With the open jet type of tunnel, the working section is not bounded by walls, but is left open. The air flows across from the nozzle, or diffuser, to the collector cone through the experimental chamber, which is airtight. The air may be drawn from and exhausted into the open atmosphere, or a continuous circulation may be provided. This conserves the energy put into the air-stream and enables greater air speeds to be obtained.

* See p. 29.

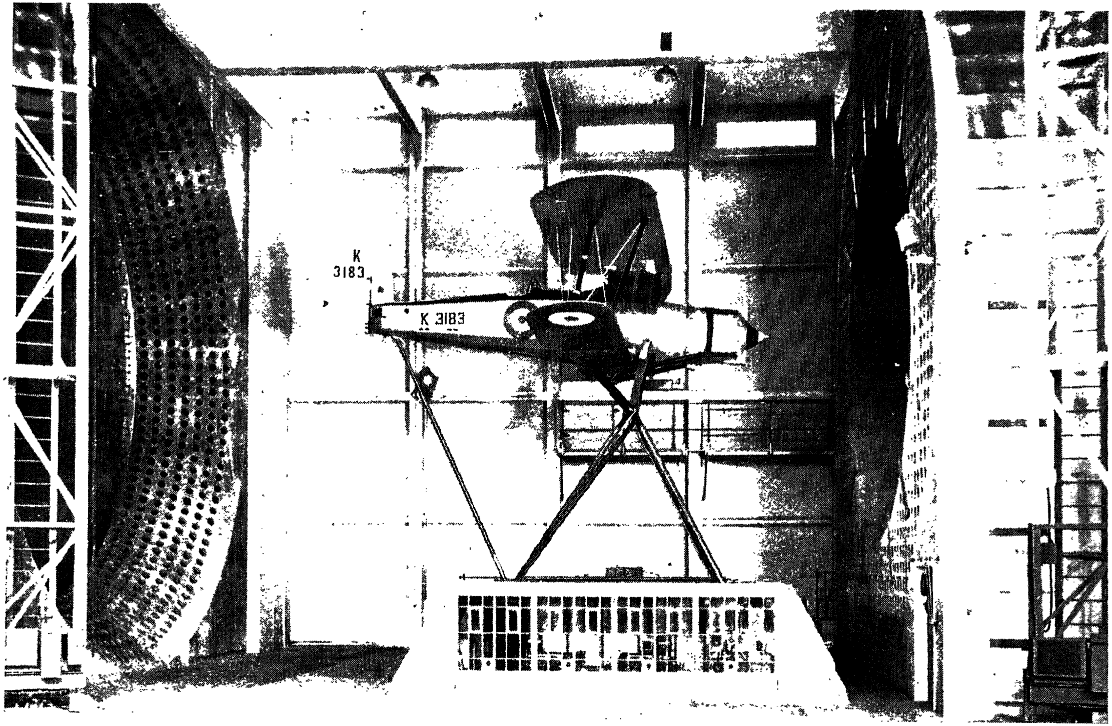


FIG. 3.—READY FOR TEST. A BRISTOL "BULLDOG" WITH NAPIER "RAPIER" ENGINE ON THE BALANCE OF THE 24FT. WIND TUNNEL

(*Royal Air Force Official Photograph—Crown Copyright Reserved*)

Facing page 10]

Open jet tunnels are used mainly for large scale work, particularly where it is desirable to make tests with complete aeroplanes, or full-size parts.

During the last few years several new wind tunnels have been installed in this country and abroad, in which air speeds approaching 300 ft. per sec. are obtainable, and, in the case of one American tunnel, full-sized aeroplanes may be tested.

The American tunnel referred to has an open jet of 60 ft. by 30 ft. cross-section, propelled by 6,500 horse-power; and produces an air-stream of 169 ft. per sec. velocity. The largest in this country is the 24-ft. diameter open-jet wind tunnel at the Royal Aircraft Establishment, Farnborough, in which a maximum speed of 176 ft. per sec. is obtainable by the employment of a 2,000 h.p. electric motor.

Wind Tunnel Balance.

The model under test is mounted on a spindle connected to a balance, by which the force acting on the model may be measured. Fig. 4 shows diagrammatically the essential features of a simple

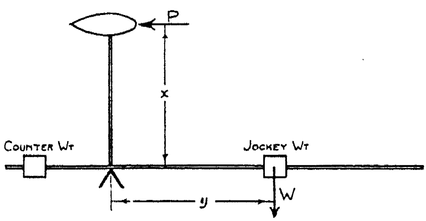


FIG. 4.—WIND TUNNEL BALANCE

form of balance. The vertical spindle passes through a hole in the tunnel base and is connected to a horizontal lever which is balanced on a pivot and is provided with a sliding jockey weight. The air force, P, acting on the model, multiplied by the vertical distance x, is equal to the product of the weight, W, and the horizontal distance y. Since the sliding weight remains constant, the lever arm may be calibrated to read directly, in lb. weight, the force on the model. For $Px = Wy$, or $y = \frac{x}{W} P = C\,P$, where $C = \frac{x}{W}$ and is a constant value.

The simple balance described is capable of measuring forces parallel to the tunnel axis only, but may be used for force measure-

ment perpendicular to the tunnel axis by horizontal rotation of the balance through an angle of 90°, or a second lever may be attached to the former at the spindle joint to allow measurements along two mutually perpendicular axes concurrently.

Counter-weights are supplied for balancing the weight of the lever arms, and to allow for any unbalanced couple that may be present due to the model not being supported at its centre of gravity. Also a vertical arm may extend downwards into an oil dash-pot for damping out oscillations. A torsion rod may be incorporated in the supporting system for measuring the torsional forces present.

Interference Effects.*

Since the artificial air-stream is bounded by tunnel walls, or by the free air surface of an open jet, it is inevitable that some constraint of the flow takes place, which in turn must have some effect on the model under test. The interference may be minimised by the use of very small models relative to the tunnel cross-section, although it is desirable for other reasons that large models should be employed.

If a relatively large model is placed in the tunnel the effects are twofold: First the air-stream is not capable of free lateral expansion as would be the case in normal flight, and secondly the area through which the flow takes place is restricted, with consequent increase of velocity. The result of the former factor is that the air-flow is restrained from following its natural pattern, and some modification of the air forces set up must result, whilst the latter factor tends to intensify the air forces, at least over some portions of the model.

In the case of an aerofoil, lift is obtained by a general downward movement of the air-flow as it passes the wing. The presence of the walls, together with the frictional resistance close to the stationary sides, prevent the full development of such behaviour and thus give misleading results, whilst at large incidence the tunnel wall and the increased velocity between the leading-edge and wall may delay the manifestation of stalled conditions.

The Compressed Air Tunnel.

In most of the older tunnels, the Reynolds' number for

* See *R. and M.*, 1566, "Wind Tunnel Interference on Wings, Bodies and Airscrews," by H. Glauert, F.R.S.

models under test has usually been roughly one-tenth that of actual flight, and therefore it is seen that full-scale conditions could be closely resembled by increasing the air density to 10 atmospheres.

This, then, is the purpose of the compressed air, or variable density, tunnels, of which type two are in existence to-day. The first was built in America with a pressure of 20 atmospheres and a working air speed of 74 ft. per sec. across a jet of 5 ft. in diameter, whilst the second is to be found at the National Physical Laboratory, Teddington. This latter has a pressure of 25 atmospheres with a speed of 90 ft. per sec. and a jet diameter of 6 feet.

The higher pressure, than the ten atmospheres indicated above, enables the use of smaller models and lower air speeds.

The results obtained by means of the compressed air tunnels have not shown the degree of accuracy at first anticipated, there being considerable discrepancy due to turbulence of the air-stream. The result of such turbulence has been found to have the same effect as an increase in the Reynolds' number, so that a test carried out in a variable density tunnel at a given calculated Reynolds' number gives results appertaining to a Reynolds' number of 2, or even 2½, times as great. In brief, the effect of turbulence is an increase in the apparent air resistance.

In the case of an aerofoil the effect of turbulence is to increase both the minimum drag and maximum lift.

The turbulence associated with the full-scale tunnels has been found to be of the same order of the turbulence of free air, and no correction factor of this nature is necessary.

An alternative method of carrying out tests at lower kinematic viscosity, or higher density, is to employ water, having a ν value of one-thirteenth that of air, as the medium instead of air, and this has been used to some small extent. Thus a model to a scale of one-thirteenth full-size could be tested at the correct Reynolds' number by being suspended in a stream of water having the same speed as for the full-sized aircraft. The difficulty in this method lies in the fact that the forces to be measured are proportional to the density of the medium and are therefore roughly 800 times as great as in the case of air. Models have to be strongly built to withstand such large forces, and all supports and measuring apparatus have to be built in like proportion.

The Vertical Tunnel.

The latest development in wind tunnel work is the vertical tunnel at the Royal Aircraft Establishment, in which the air-flow is vertically upwards, for use in investigating the problem of spinning.

The model under test can be caused to descend in spinning flight, and by adjustment of the air-flow the model can be kept at a constant height, and thus both examination of the phenomenon and photography are greatly facilitated.

The tunnel mentioned has a diameter of 12 ft., is 30 feet high and has an air speed of 30 ft. per sec.

Chapter II

AIR-FLOW AND STREAMLINES

Streamlines and Resistance.

It has already been stated that the presence of an object causes a change in the steady flow of air. Both the course and speed may be affected. Reference was also made to the idea of regarding air-flow as forming laminæ, the courses followed by the layers being termed streamlines, since they show the flow direction of the air particles along the course. Objects which cause little upsetting of the steady flow of air are said to be well streamlined, or merely streamlined.

Fig. 5 has been prepared from photographs depicting the flow of a fluid past different shaped objects. The medium employed in this case was water, the streamline effect having been obtained by means of aluminium dust sprinkled on the surface. In the case of a flat plate, held perpendicularly to the original direction of flow, the fluid is seen to bifurcate and pass out to the edges of the plate. Between the inner boundary of the forward flow and the plate there is a cone of more or less stagnant fluid, or, more correctly, the fluid within this region is subjected to slow eddying motion, but with some leakage past the plate edges, which is compensated for by the introduction of fresh fluid.

Behind the plate the flow is not able to close in again with such ease, with the result that turbulence is established. Fluids may be caused to follow converging, or receding, surfaces, provided abrupt changes of direction are avoided, but so soon as the necessary inward acceleration, in the direction normal to the surface, exceeds a certain amount, there is a refusal on the part of the fluid to comply. The flow is then said to *break away*, or separate, and is accompanied with eddy formation.

This is the fundamental principle underlying most of the problems in aerodynamics, and, simple though it appears, its importance cannot be too highly stressed.

Reverting now to the flow pattern round the flat plate, it is

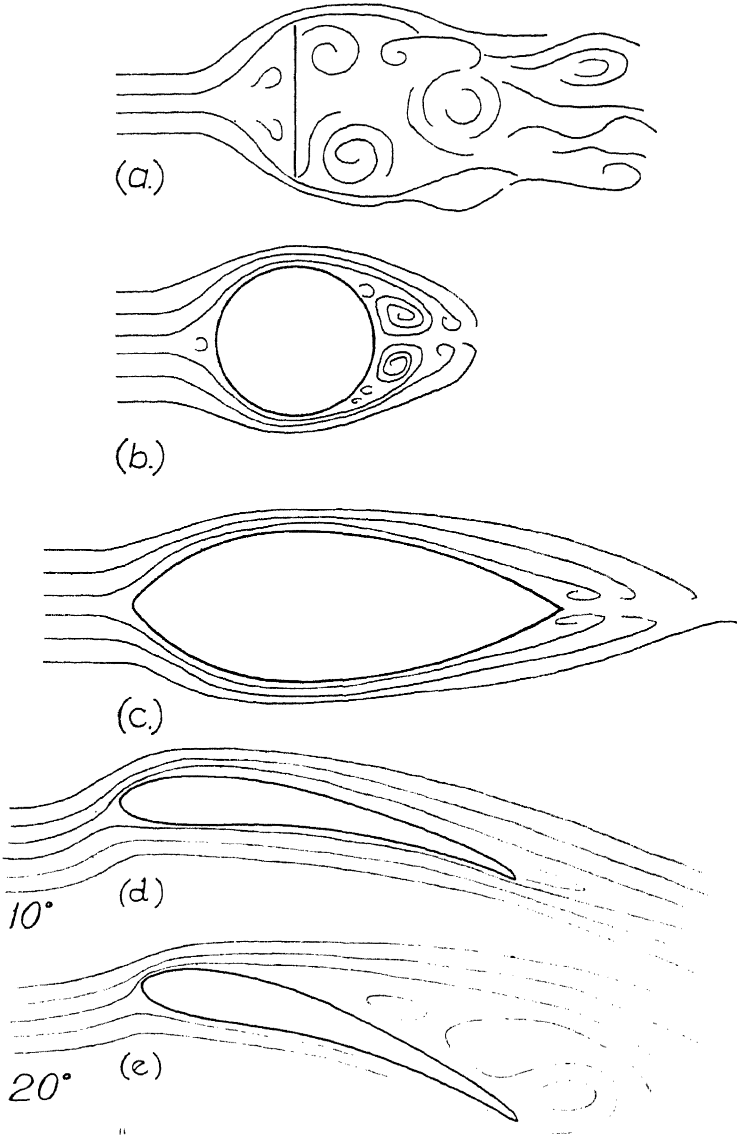


FIG. 5.—FLUID FLOW PAST BODIES

seen that the fluid at the edges has a considerable outward direction of motion, so that in order that it might converge again

on the down-stream side, a very large angle of turn would be necessary in a short space of time; in other words the inward acceleration would be very great. In the case considered the necessary acceleration is excessive, and so turbulence is set up.

The eddies build up in the form of cylinders, or rolls, just behind the edges, which enlarge, break away, and move off down stream, when their places are taken by new vortices in the initial stage of formation. Thus a trail of rotating eddies is formed behind the edges. The vortices form alternately on opposite edges, with a distinct period depending on the velocity of flow and other factors.

The flow past a sphere is shown at (*b*), in which the general features resemble those of the flat plate, but, the tendency for the streamlines to come together after passing being greater, there results a smoother flow, and the resistance set up by the presence of the sphere is only about one-tenth that for the plate.

It is noticed that a strong eddy-forming influence still exists behind the sphere and this can be partially reduced by the provision of a conical tail, thus reducing the drag by one-half. At (*c*) is shown a shape approximating to a true streamline form, for which the eddy motion has been almost eliminated, and thus enables the surprisingly low drag figure of little more than 2 per cent. of the flat plate to be obtained. Such is called a streamline shape.

Figs. (*d*) and (*e*) show the flow of a fluid past the section of a wing at small and large angles of incidence, from which it will be noticed how the flow tends to depart from the purely streamline as the angle is increased. The eddy formation, which starts at the trailing-edge, grows with increased incidence and spreads forward over the rear part of the top surface, and it will be seen later that this phenomenon is one of the most important aspects of flight and one of the most difficult the aeroplane designer and pilot alike have to contend with.

Flow Patterns of Wool Streamers.

A quick, simple method of observing air-flow over wings and other bodies is provided by attaching a number of wool streamers to the forward part of the body and observing their behaviour. The flow is sometimes explored by means of a streamer attached to a light rod which may be manipulated from outside the wind tunnel. These streamers do not, however, show the true stream-

lines, since their own weight tends to lower the free end, and also the resistance of the wool tends to straighten the streamer and mislead the observer. These defects may be largely overcome by the use of a larger number of short streamers, or by their attachment to a movable support.

The next figure has been obtained by the method just explained, the air speed at which the test was made being 60 ft. per sec. Local instability of flow is shown by dotted lines, and where necessary the direction of the flow has been indicated by arrows.

The small diagram of Fig. 18 shows how the air-flow is noticed to oscillate between the two flow patterns (*a*) and (*b*) at large incidence. This feature of the flow is similar to the eddy formation behind a flat plate, already mentioned, in which the eddies form alternately at each edge. The phenomenon known as tail buffeting, which takes place at high incidence of the main plane, is obviously closely related to this state of flow.

Flow Patterns by Smoke Trails.

The smoke tunnel is a miniature wind tunnel in which smoke is introduced just forward of the model for observing the behaviour of the stream. Owing to the slow speeds to which the smoke tunnel is limited, the flow patterns cannot be regarded as being truly representative of full-scale conditions, but nevertheless, much can be learnt by smoke observation and unexpected phenomena may be quickly explained by this means. Fig. 7 shows the flow patterns of a number of objects obtained by the smoke tunnel. Where the smoke becomes diffused, denoting turbulent conditions, the diagrams have been shaded : Where a smoke trail remains clearly defined, but changes its path, like the moving course of a water-stream, the successive positions are marked in ; the general direction of mass movements, as compared with individual trails, or streamlines, being shown by arrows in the diagrams.

Skin Friction.

If a plate of negligible thickness is held edgewise in a moving stream so as to cause no eddying, or change of flow direction, the whole of the resistance experienced is skin friction, and is due to the slowing down of the fluid close to the surface by its own viscosity.

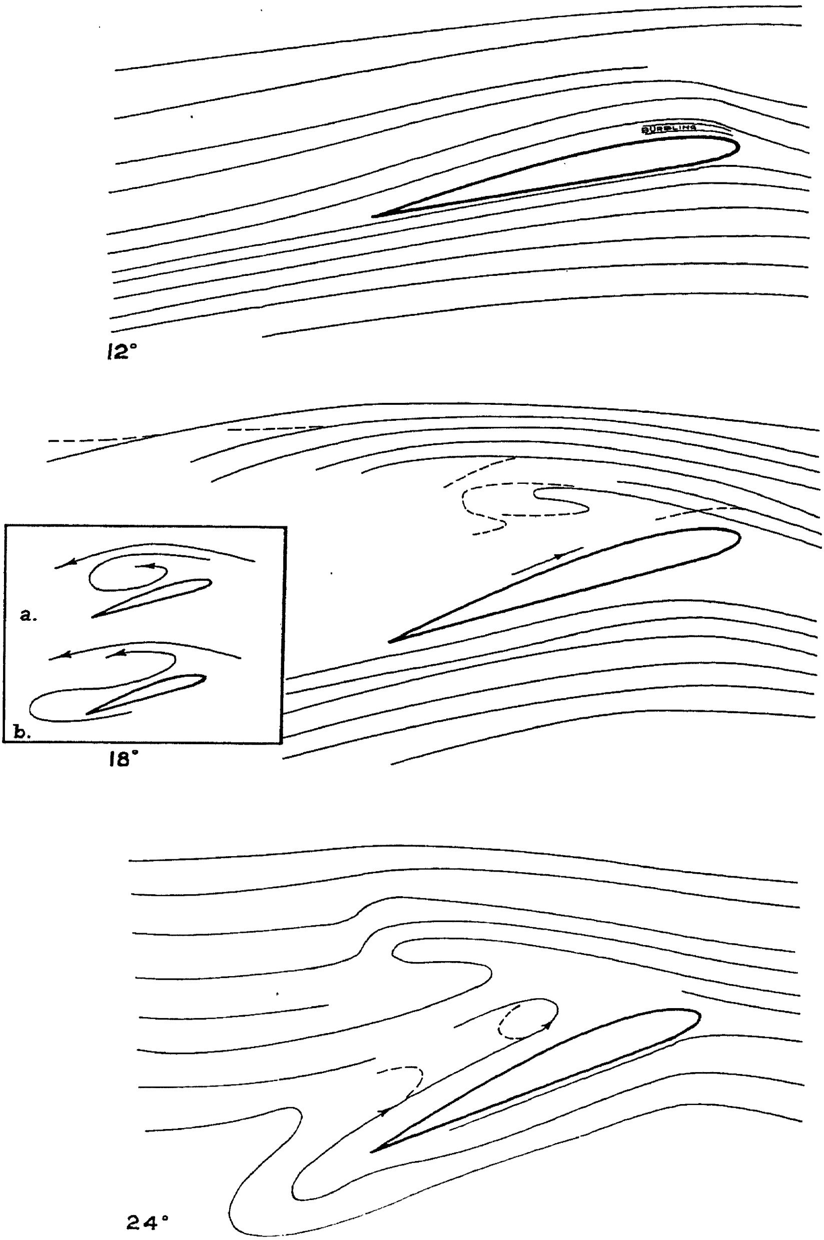


FIG. 6.—AIR-FLOW PLOTTING BY SERIES OF STREAMERS

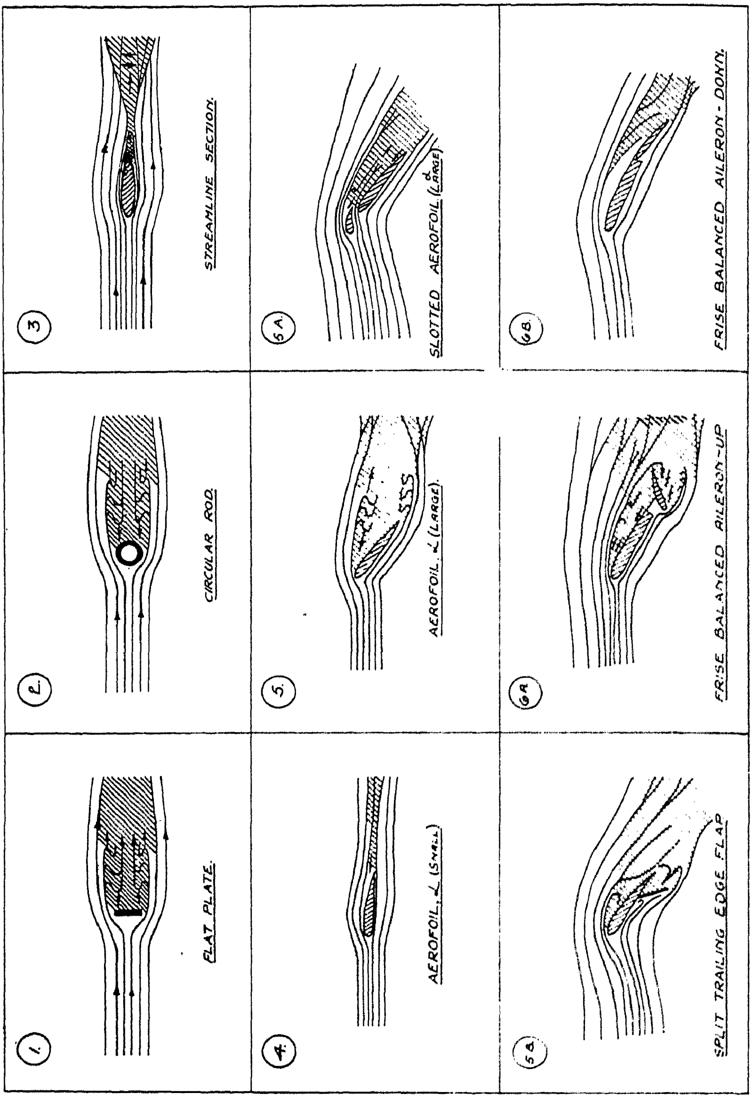


FIG. 7.—SMOKE PATTERNS

BOUNDARY LAYER.—The fluid immediately adjacent to the surface has the same velocity as that of the body, i.e., in the case of a stationary body the fluid stream in its close vicinity is also stationary. Each thin lamina of fluid acts on its bounding lamina so that a velocity gradient is formed with increase of velocity away from the surface.

The boundary layer is defined as that depth of the stream adjacent to the surface of a body in which the velocity is not uniform. Fig. 8 (*a*).

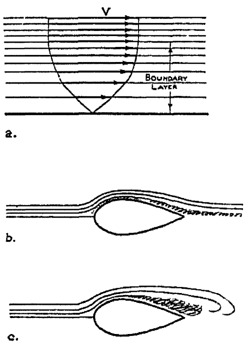


FIG. 8.—BOUNDARY LAYER FLOW

The boundary layer thickness is least at the nose, or leading-edge, of the body, and builds up towards the rear, proportionately to the square root of the distance from the leading-edge.

The thickness is also a function of velocity, and decreases in inverse proportion to the latter. This is a point of importance in connection with the varying resistance of bodies with change of Reynolds' number.

So far the boundary layer has been assumed to consist of laminar flow, but it may also consist of turbulent flow, in which the fluid particles have additional accelerations normal to the surface.

It is usual for the flow within the boundary layer to remain laminar over the front part of the surface, after which it becomes turbulent. The point of transition from one form to the other is difficult to determine accurately, and is dependent on the smoothness of the undisturbed stream, surface texture, and on the velocity. Fig. 8 (*b*).

The depth is generally very small, say, some fraction of an inch, but depends on body shape, position on body's surface, degree of initial smoothness of the fluid flow, velocity of flow, and surface roughness. Over the nose of an aerofoil the depth is perhaps a few thousandths of an inch, but in the case of wind over the earth's surface the boundary layer is said to reach a thickness equal roughly to 40 times the wind speed per second.

The shearing between lamina and lamina, within the so-called boundary layer, constitutes of course skin friction.

In general a laminar boundary-layer flow gives less resistance than a turbulent flow,* but it will be seen later that under certain circumstances the reverse may be true.

Over the forward part of a body the streamlines diverge, and, of course, suffer an increase in velocity, with an accompanying pressure drop. At, or near, the widest section of the body

* See p. 187

the pressure increases again, and the particles in the slow-moving boundary layer, devoid of the necessary kinetic energy, tend to pile up, with a thickening of the layer, which in turn tends to deflect the streamlines beyond the boundary layer into the adjacent stream, until it in turn becomes turbulent, breaking away from the contour of the surface, and giving rise to a system of vortices with greatly increased resistance. This is termed *separation*. Fig. 8 (*c*).

The effect of initial turbulence in the fluid stream is to delay the advent of separation. This is due to the energy present in the turbulent boundary layer, which helps to prevent stagnation. This is a factor of great importance, and explains why tests in different wind tunnels are not always in good agreement, and why the results obtained in compressed air tunnels, where considerable turbulence is present, sometimes show higher values of the aerodynamic characteristics than other wind tunnel, and full-scale, test results.

It is worthy of note that slight roughening of surface tends to cause turbulence of flow, and, for the reason just given, a lowering of resistance often results, though at high Reynolds' numbers a smooth surface is essential for low resistance. A practical application of this knowledge is the beneficial resistance effect obtained by a small degree of surface roughening of small, exposed parts of aircraft at all but very high speeds.

Recent research has shown that when the boundary layer is turbulent there exists a sub-layer, of very small depth, adjacent to the surface of the object, in which the flow remains laminar. The discovery of this sub-layer is most important in considerations of drag, and its thickness in inches is given within reasonable accuracy as $T = \frac{1}{6{\cdot}7\ V}$, where V is the velocity in ft. per second of the air-flow at the point under consideration.

Resistance.

It has been seen that the resistance of a body to forward motion, or profile drag, depends principally on its shape and is made up of two components; form drag and skin friction. Form drag is due to the formation of eddies, that accounts for almost the whole resistance of a flat plate placed normal to the relative flow, together with the resultant of tangential pressures over the body surface, generally of small account. Skin friction

accounts chiefly for the drag of a flat plate when placed parallel to the flow. With most bodies both resistance components are present in varying degree, depending on the body shape, but it has been seen that eddy drag is almost non-existent with good streamline forms, and the suppression of this component to the absolute minimum is the aim of the aeroplane designer.

It has already been seen that the resistance to motion of any body is given by:

$$R = C_R \frac{\rho}{2} S V^2,$$

where C_R is the coefficient of resistance, depending on the shape of the body, or as we now see, on the components of skin friction and eddy formation. The skin friction component may be calculated with a fair degree of accuracy, the coefficient* value varying between 0·002 and 0·008, according to the Reynolds' number,† but the drag resulting from the generation of eddies is not easy to determine directly, and instead therefore the results of wind tunnel tests are relied upon for the total resistance coefficients.

THE EVOLUTION OF THE AEROFOIL OR LIFTING SURFACE

Flat Plate.

Most of the consideration so far has been restricted to bodies having their axis of symmetry parallel to the direction of motion, in which case their resistance has acted directly against the force producing motion. Suppose, however, the flat plate to be inclined at some angle to the direction of motion, Fig. 9. The resistance force now acts approximately at right angles to the surface, whilst the relative air-stream is directed downwards to some extent. Thus there is a tendency for the plate to be lifted, as well as having its forward motion, relative to the stream, retarded.

The total reaction, R, may, for convenience, be split into components perpendicular and parallel to the direction of movement. These components are termed *lift* and *drag*, and are denoted by L and D respectively, or if α is the angle of inclination

* $C_{Df} = 0{\cdot}67\left(\frac{vl}{\nu}\right)^{-0.5}$ for laminar flow, and $0{\cdot}074\left(\frac{vl}{\nu}\right)^{-0.2}$ for turbulent flow.

† See p. 211 and Fig. 130.

of the plate, or, more correctly, the angle made by R with the vertical, then :—

$$L = R \cos \alpha, \text{ and } D = R \sin \alpha.$$

$$\text{But } R = C_R \frac{\rho}{2} S V^2,$$

$$\therefore L = C_R \frac{\rho}{2} SV^2 \cos \alpha$$

$$= C_L \frac{\rho}{2} S V^2, \text{ where } C_L = C_R \cos \alpha,$$

a constant at any particular value of α.

C_L is the coefficient of lift, and equals $\dfrac{L}{\frac{\rho}{2} S V^2}$ . . . (4)

Similarly $C_D = \dfrac{D}{\frac{\rho}{2} S V^2}$ . . . . (5)

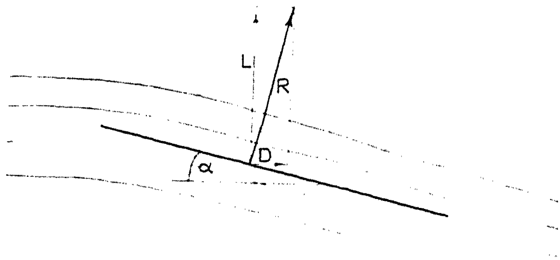


FIG. 9

The smooth flow conditions of Fig. 9 hold good for small angles of the plate only, above which the air-flow is no longer able to conform itself to the shape of the plate, so that the streamline flow breaks down, and gives place to eddy formation (Fig. 10 (*a*)).

Curved Leading-Edge.

If the leading-edge of the plate is curved downward, so that the forward part of the plate lies along the direction of motion, whilst the rearmost portion remains as before, the air is able to follow the gradually changing contour, and streamline conditions are again obtained (Fig. 10 (*b*)). In this way it becomes possible to deflect the air downwards through a larger angle, and thus to

obtain greater lift values than were attainable with the flat plate. Unfortunately, a limiting value is again reached, beyond

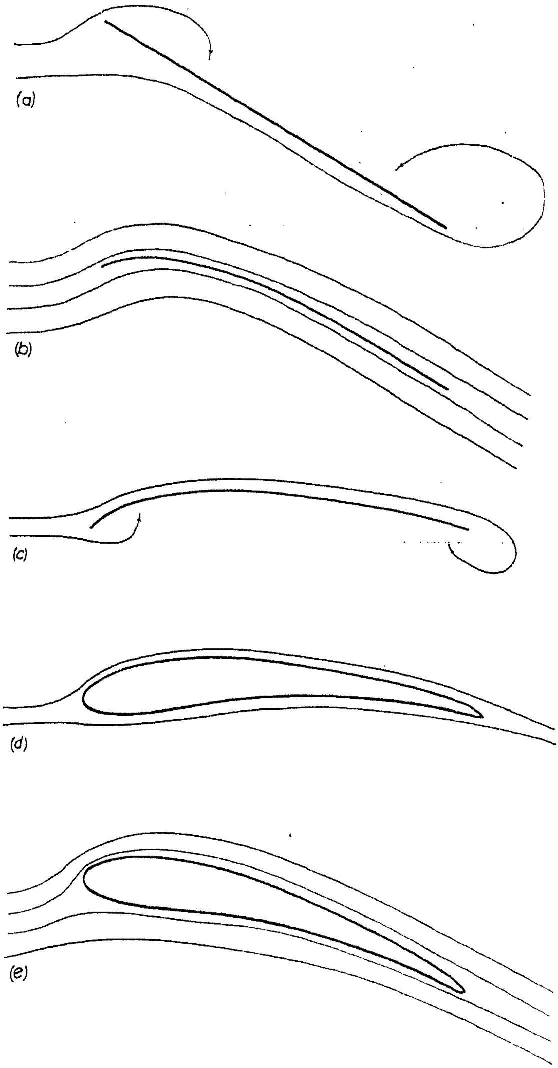


FIG. 10.—EVOLUTION OF THE AEROFOIL

which stalled conditions set in once more. The close relationship between this problem and that of the flow of a fluid past a sphere and streamlined shapes is readily apparent. Once again the

critical point is reached where the air, or other fluid, is unable to close in to the receding surface with a smooth flow.

The shape so far evolved is suitable for flight at the larger end of the angular range, but has now become inefficient at small angles of incidence, on account of the eddy motion created below the forepart of the aerofoil (Fig. 10 (*c*)), and this is next overcome by filling in, or reducing the concavity of, the underside, which thus results in a shape that is efficacious throughout a comparatively large range of angles, and moreover provides suitable housing space for the main strength members, or spars (Fig. 10 (*d*) and (*e*)).

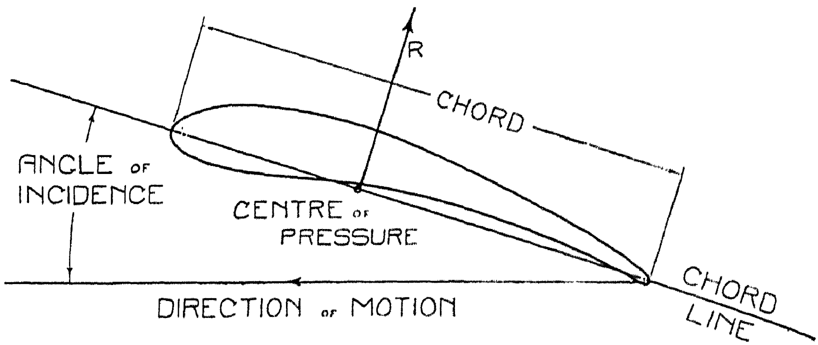


FIG. 11

It only remains now to find the curvatures best suited to the various flight and structural requirements, this being done chiefly by means of extensive wind tunnel experiments.

The next diagram shows the terms by which the physical qualities of an aerofoil are known. The angle made by the chord line with the direction of motion, or relative air-flow, is called the *angle of incidence*, or *angle of attack*. The *chord line** is the line joining the centres of curvature of leading- and trailing-edges, and the *chord* is the length of the chord line between the intersection with the surface at the leading- and trailing-edges.

The *centre of pressure* is the point on the chord line through which a single force would act, if the total air reactive forces were replaced by a single force to give the same effect.

* The aerodynamic characteristics of many aerofoil sections in published reports are referred to datum lines which touch the under-surface at two points.

CHAPTER III

THE ATTAINMENT OF LIFT

IT has been seen that in order that an aeroplane may be supported in flight it is necessary that some other mass be given a downward acceleration, and this is accomplished by driving a plate, or aerofoil, through the air inclined at some small angle to the direction of motion. A similar example is provided by the surf-board, of 3 or 4 sq. ft. area, which is dragged over the surface of the water, also at some small inclination, and is thereby able to support the weight of one, two or even three persons. It may be said, then, that in general the lift of an aircraft is obtained by the impartation of downward momentum to the air, and this is illustrated by the series of diagrams of Fig. 12. Unlike most flow patterns, which show the movement of the fluid past an object, there being no relative movement between the camera and object, these have been obtained by giving the camera and fluid the same uniform translational motion, and photographing the fluid displacement caused by the passage through it of the body.

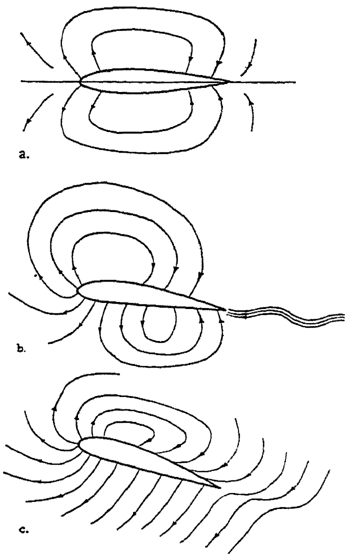


FIG. 12.—FLOW PATTERNS

At (*a*) is shown the flow induced in an otherwise stationary fluid, by the passage of a streamline body having its axis parallel to the direction of movement. It is noticed how the nose of the body drives the fluid forward and outward, with the subsequent closing in again at the rear. The second diagram (*b*) shows the flow set up by an aerofoil at a small angle of attack. The fluid below the leading-edge is seen to flow forward and upward;

a tail, or wake, having a forward motion denotes drag, while the downward general tendency of the particles shows that lift is being generated.

Diagram (*c*) is the same aerofoil section at a large angle of incidence. The downward and forward movement shows that lift and drag reactions are present on the aerofoil. The forward flow of the wake is also again noticed.

Lift due to Change of Momentum of Air Mass acted upon.

In Fig. 13 let the two streamlines indicate the paths of two particles of air which pass above and below the aerofoil. In its

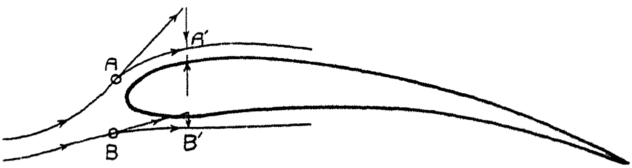


FIG. 13

path from A to A′ the upper particle has been deflected from its initial course in a downward direction. The small force producing this downward acceleration must be accompanied by an equal and opposite reaction which tends to lift the aerofoil to meet the particle. Similarly the lower particle in moving from B to B′ is displaced vertically, as shown, and likewise a force is produced which tends to push the aerofoil upwards.

It follows from the above that when the air-flow bends outward away from the surface, the pressure is increased but if the flow is constrained to move out of its path towards the surface, then a depression is produced. This is illustrated in the next diagram and inspection of pressure diagrams will show later how the pressure distribution is affected in such a way.

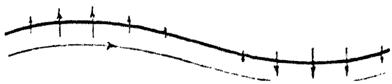


FIG. 14

Furthermore the change in pressure will be greatest where the curvature (of the air-stream) is greatest, provided that it is not carried to such extent that the air-flow breaks away, and thus causes turbulence to be set up.

Relation between Pressure and Velocity.

If a stream of fluid is flowing along a pipe, or other conduit, in streamline flow, its total energy, apart from frictional losses, must remain constant, unless it is acted upon by some external force. The energy is composed of pressure, momentum and height energy, so that variation of one may take place only at the expense of one, or both, of the other components. This is known as Bernoulli's theorem. In the present consideration the effect of height can be neglected if the flow is supposed to be horizontal, or sufficiently so to render its effects negligible. There remain pressure and velocity, and these are connecte the law

$$p + \tfrac{1}{2} V^2 = C, \text{ where C is some constant, or}$$
$$\text{pressure } p = C - \tfrac{1}{2} V^2.$$

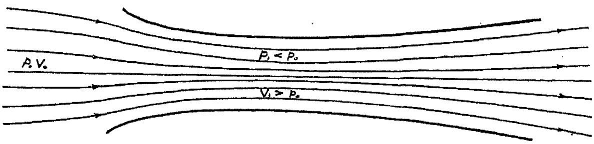


FIG. 15.—VENTURI TUBE

Suppose now the flow considered meets with some restriction, so that the cross-sectional area of the pipe is decreased, it naturally follows that some speeding up of the flow must take place at this point, and in consequence the pressure must decrease.

It is thus seen that if V is made to increase, p must decrease, and conversely if V decreases, p must increase. Or where streamlines close in, a decrease in pressure is indicated, and *vice versa*.

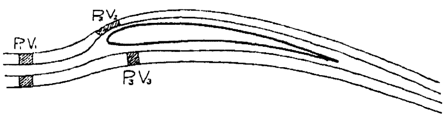


FIG. 16

Consider the application of this to an aerofoil. Fig. 16 shows the streamlines denoting flow past a wing section under conditions of smooth flow, from which it is noticed that a "bottle

neck," or throat, occurs just above the leading-edge, whilst there is an increased flow area immediately below, both of which are indicated by the spacings of the streamlines. Hence there is a loss of pressure above and a gain of pressure below, or $p_3 > p_1 > p_2$, from which $p_3 > p_2$. The difference between the pressures, i.e., $p_3 - p_2$, is the resultant upward pressure causing lift over the parts considered.

It will also be noticed that the lowered pressure over the top surface near the leading-edge, and the increased pressure below, together create an upward tendency of the flow just forward of the leading-edge. This is an important feature of the air-flow patterns of aerofoils and is termed the "upwash." The upwash may be perceptible up to a distance two chord lengths in front of the leading-edge.

Before passing it may be mentioned that the pressure effect just discussed also takes place to a lesser degree near the trailing-edge, but this time in the direction opposite to that desired, and this does in fact cause a decrease in the total lift.

Circulation Theory of Aerofoils.

The length of path followed by the air passing over the upper surface of an aerofoil is greater than the path of flow below, assuming smooth flow conditions to hold good (See Fig. 16). It would therefore be expected that the average

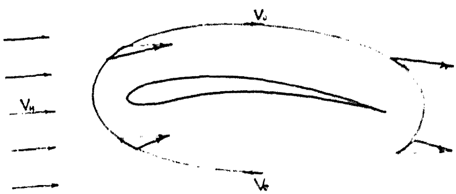


FIG. 17.—CIRCULATION THEORY OF AEROFOIL

velocities over the two paths would be unequal, and consequently the pressure below should be greater than the pressure above.

It would be possible to obtain the same set of conditions if a rotational motion is superimposed on a steady air-flow so that the velocity from leading-edge to trailing-edge is increased over the top, and decreased over the lower surface. That such a rotational flow component does actually take place is

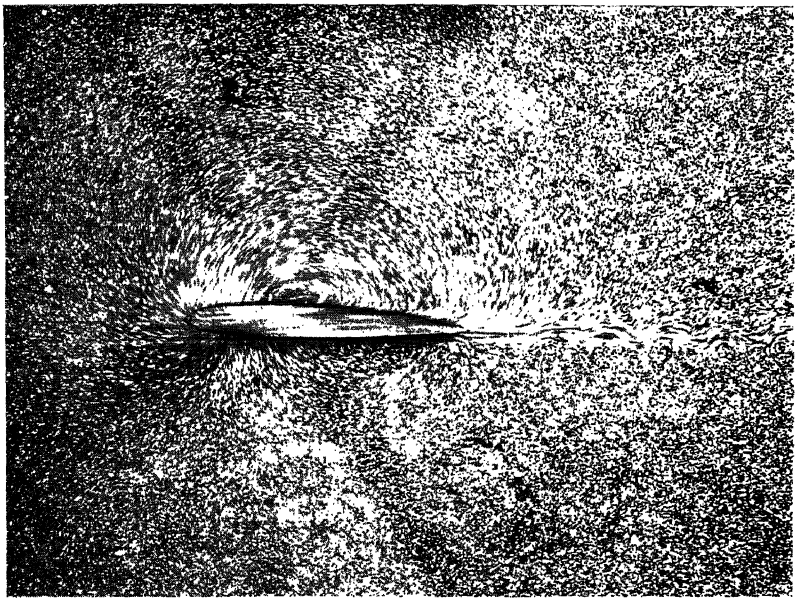


FIG. 18.—CIRCULATORY FLOW AROUND AEROFOIL. INCIDENCE SMALL

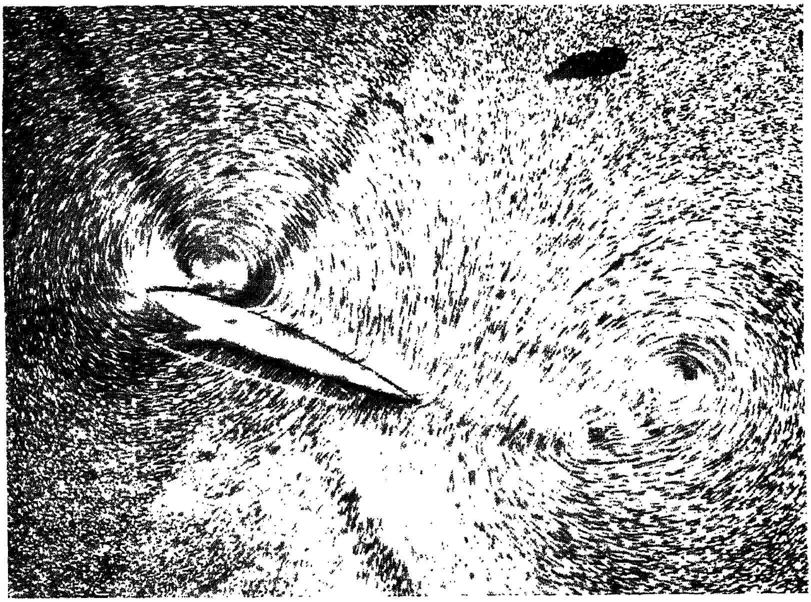


FIG. 19. CIRCULATORY FLOW AROUND AEROFOIL. INCIDENCE LARGE

[Facing page 30

shown by an inspection of Fig. 12 (*c*). This is known as the circulation theory of lift.

It is easy to reconstruct the more usual flow pattern from the one depicting circulatory flow (Fig. 12 (*c*)) by adding horizontal flow components, V_H, to the circulating flow V_C, and this has been done diagrammatically in Fig. 17 above, in which V_C is made small relative to V_H. Four points only have been considered, but it will be noted that the resultant velocities thus found agree reasonably closely, both in magnitude and direction, with the air-flow pattern of the previous figure.

Aerofoil Velocity Diagrams.

It is of interest to see how the air velocity in the region of a moving aerofoil does vary from point to point, and this is best done by supporting an aerofoil in a moving air-stream and noting the results.

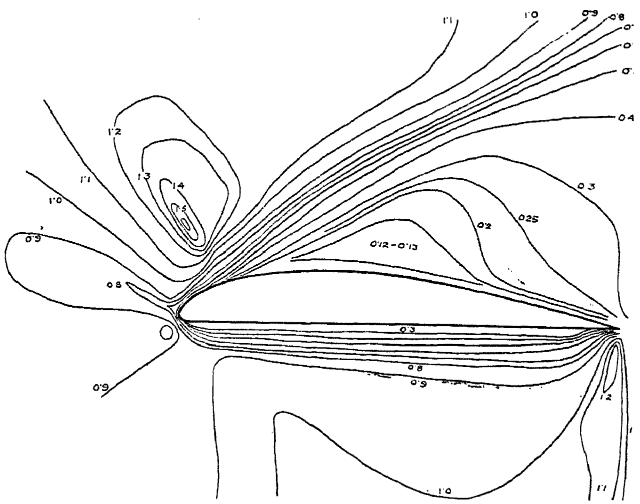


FIG. 20.—AEROFOIL VELOCITY CONTOUR DIAGRAM. $\alpha = 4{\cdot}5°$

The diagram is fairly typical of those so far obtained by exploration in the vicinity of an aerofoil with a hot-wire anemometer,* and shows by means of velocity contour lines the variation of the air velocity as it passes the aerofoil.

Above the wing the velocity increases fairly rapidly over the

* "On the Flow of Air Adjacent to the Surface of an Aerofoil," N. A. V. Piercy and E. G. Richardson, *R. & M.*, No. 1224, Dec., 1928.

first tenth part of the chord to roughly 50 per cent. above normal, after which it falls off rapidly again to as low a figure as one-eighth normal, followed by a fairly steady increase to only one-third normal velocity vertically above the trailing-edge.

Just ahead of and below the leading-edge the velocity drops to 0·7 normal, increases thence to about 0·9 for the greater part of the chord length and rises again to 1·3 just below the trailing-edge.

The venturi effect, already mentioned, above the leading-edge and below the trailing-edge is clearly demonstrated.

If this velocity pattern is interpreted to show the effect of a moving aerofoil upon initially stationary air it is seen that there is a region above the rear two-thirds in which the air is dragged forward with the aerofoil, but at a slightly lower speed. The air below the forward four-fifths of the chord, and extending well forward of the leading-edge, is also given a small velocity forward in the direction of the moving aerofoil. There are two regions, one above the leading-edge and the other below the trailing-edge, in which the air is given a considerable backward velocity. The effect of skin friction is clearly seen from this diagram.

The passage of a poorly streamlined aeroplane (locomotive, motor-car, or other vehicle) through the air, causes a considerable volume of air to be set in motion in the same direction, and this may be distinctly felt as a vehicle passes a stationary observer.

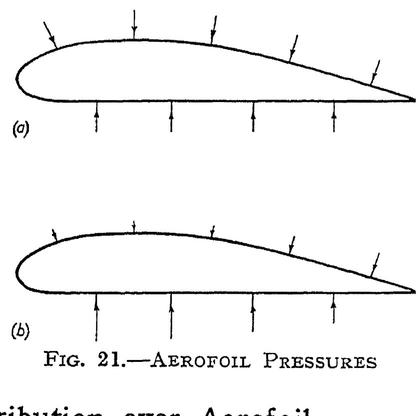


FIG. 21.—AEROFOIL PRESSURES

Pressure Distribution over Aerofoil.

A stationary aerofoil is subjected to the pressure of the atmosphere, which acts with equal intensity at all points, the direction being normal to the surface (see Fig. 21 (*a*)). The total up force

must equal the total down force, since the horizontally projected areas are the same in both cases. Similarly the horizontal forces, or components of forces, must cancel out, and so the aerofoil is in equilibrium.

If the pressure over any one part of the aerofoil is changed, the total forces will have an unbalanced resultant tending to move the aerofoil away from the region of the greater pressure intensity towards the zone of lower pressure. It does not matter whether the alteration in pressure is a decrease, say, on one face, or an increase on the opposite face, the result is the same. Also a decrease on one side accompanied by an increased pressure on the opposite side accentuates the pressure difference, and increases the tendency for the aerofoil to move.

The lifting forces on a wing are brought about, in general, by an increase in pressure on the underside and a decrease on the topside (Fig. 21 (*b*)). The difference between the vertical components of these two pressures is a measure of the amount of lift. It should, however, be noticed that there is still pressure over the upper surface, which in reality is quite considerable. (A wing loading of 10 lb. per square foot is equivalent to a vertical pressure difference of 0·07 lb. per sq. inch. An *average* upper surface pressure of, say, 0·04 lb. per square inch below normal atmospheric, and an *average* increase of 0·03 lb. per sq. in. over the under surface, satisfy these requirements.)

Briefly, then, it can be said that *lift is obtained by the difference in pressures below and above the aerofoil,* and that the average pressure difference is comparatively small. The cause of this pressure difference has already been discussed.

Pressure Measurement.

It has been seen that, if the lifting force present on a wing is known, then the average pressure difference between the top and bottom surfaces can be calculated. Actually, however, the pressure intensity is never constant across the chord, but varies from leading-edge to trailing-edge on both upper and lower surfaces. The pressure gradient changes according to the angle of incidence, as also do the points of normal pressure, i.e., where the pressure changes from below to above atmospheric. The pressure distribution can be obtained experimentally by means of a series of "U" tube manometers connected to small holes in the surface of the aerofoil under test. Fig. 22 shows diagram-

matically how this is carried out. If the pressure close to the hole in the upper surface, A, is less than that of the atmosphere, B, there will be a smaller force exerted on the liquid at C than at D, and in consequence the levels will rise and fall accordingly until the pressure due to the column of liquid, of height h, is equal to the difference in pressures at A and B.

Hence h is a measure of the pressure decrease at A, and can be transferred to pressure measurement in suitable units if desired. For example, suppose h is equal to 2 in. of water, then this is equivalent to a pressure of $\frac{2}{34 \times 12} \times 14{\cdot}7 = 0{\cdot}072$ lb. per sq. in. approximately.

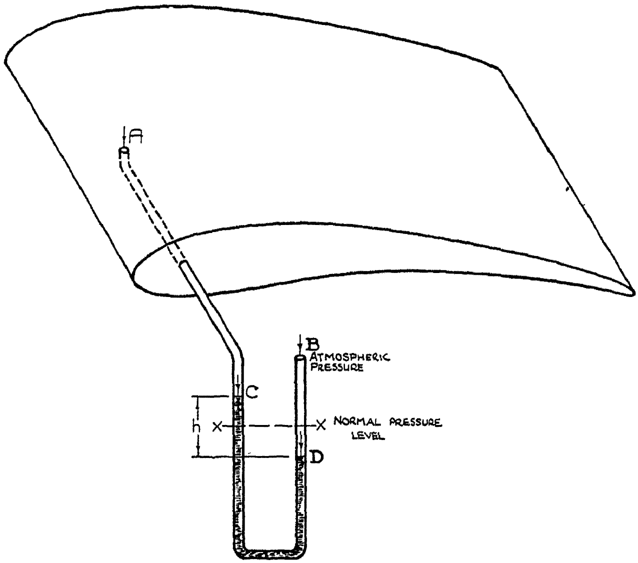


FIG. 22

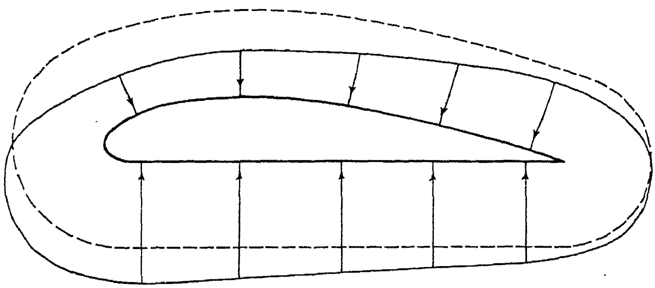
(A column of water 34 ft. high gives a pressure roughly 14·7 lb. per sq. inch.)

Thus the true pressure at A is 14·7 − ·072 = 14·628 lb. per sq. inch. (Assuming that atmospheric pressure is 14·7 lb. per square inch.)

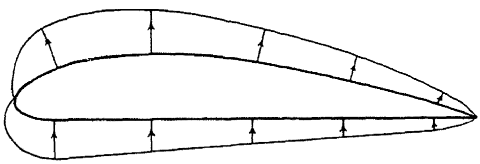
By employing a large number of manometer tubes, similarly arranged, the pressure variation over the whole surface may be obtained and plotted, as shown by the full line on Fig. 23 (a). The similarity between this and Fig. 21 (b) is noticed. If the atmospheric pressure is plotted also, as indicated by the dotted

curve, the difference between the two curves gives the alteration in pressure.

It has been seen that the actual pressure difference is quite small in relation to atmospheric pressure, and could not, in fact, be conveniently plotted in the form of this diagram, and for this reason it is usual in practice to plot the pressure *difference* only. This has been done at (*b*), Fig. 23. The upper curve denotes pressure *below* atmospheric, whilst the lower curve shows the pressure increase *above* atmospheric.



(*a*)



(*b*)

FIG. 23.—AEROFOIL PRESSURE DIAGRAMS

It will be noticed that in the second diagram the arrow heads within the negative pressure region are shown pointing away from the aerofoil surface, which may be deceptive unless the meaning is properly understood, for it has been seen that the absolute pressure at all points on the external surface of an aerofoil is positive.

Pressure Diagrams.

Pressure distribution diagrams are shown in Fig. 24 for three angles of incidence. At (*a*) the aerofoil is at, or near, the no-lift angle, for which it is seen that the relative air-flow " meets "

the wing at, and just above, the leading-edge, this being the point of bifurcation of the flow. The pressure intensity is increased over this portion, but is decreased again to below normal on both upper and lower surfaces where the air-flow is being turned inwards again.

At a moderately large angle of attack, (*b*), the pressure diagram is more conventional, and shows an increased pressure on the underside, with decreased pressure above. It is noticed that the greatest pressure differences take place over the forward part of the aerofoil and it is in this region that the streamlines undergo the greatest change of direction.

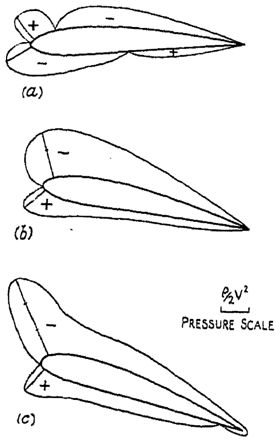


FIG. 24.—AEROFOIL PRESSURE DIAGRAMS

A point of interest at large angles of attack, (*c*), is the reversal of pressure that takes place on the underside near the trailing-edge. This is due, of course, to the strong venturi effect, already mentioned, and is accompanied by the air flowing round to the top side from underneath.

It will also be noticed that the pressure difference at more normal angles (say 10°) is greater above than below the aerofoil; the proportion being roughly two to one.

Dealing with the actual pressure values, the scale for each diagram being given in terms of $\frac{1}{2} \rho V^2$, it may be stated that negative pressures of four times stagnation pressure are sometimes reached at high angles of incidence, the point of maximum negative pressure being on the upper surface just behind the stagnation point. This pressure represents a loading of 25 lb. per sq. foot at 50 m.p.h., and 100 lb. per sq. ft. at 100 m.p.h. As would be anticipated* the maximum positive pressure under all conditions takes place at the stagnation point, and is, of course, $\frac{1}{2} \rho V^2$.

Each pressure curve may be analysed into lift and drag curves, by plotting the vertical and horizontal pressure components. The areas within these curves give the lift and drag values respectively.

* See p. 7.

Centre of Pressure.

A definition of the centre of pressure has already been given. It is convenient for many purposes to assume that the total pressure forces across the chord of a wing act as a single force, the point of application being then termed the centre of pressure (C_P). It has been seen that the nature of these pressures, or forces, changes for different angles of attack.

At a high incidence (about 15°), the C_P is in its most forward position, roughly $\frac{1}{4}$ to $\frac{1}{3}$ of the chord from the leading-edge for most aerofoils (known as the C.P.F. position), and this is only as would be expected from an examination of the last figure. At small angles the C_P moves towards, and may even go beyond, the trailing-edge. At first this may appear confusing, because it seems unlikely that a pressure acting over the surface of an object could have its centre right away from the object. Reference to Fig. 24 (*a*) shows that the pressure regions give rise to a down-

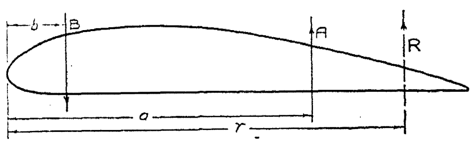


FIG. 25

wardly acting force over the front and an upward force over the rear of the aerofoil, shown as B and A in Fig. 25. The total reaction is R, such that $R = A - B$, that is R is less than A, and hence its *assumed* point of application may be further back than A: for, by moments about the leading-edge $R \times r = A \times a - B \times b$, and if A and B are fairly similar in magnitude, but b is small compared with a, then r will be large.

The C_P position is thus seen to be imaginary, although the twisting effect on the wing due to the pressure distribution is correctly given by assuming the resultant force to act through such a point. It will also be realised that for this condition *the total pressure forces acting on the wing are greater than the nett reaction* R, and consist of a down load over the fore part, together with an up load over the rear.

For top speed, horizontal flight, the C_P is generally near the mid-chord position. This is known as the centre of pressure back (C.P.B.) position.

Centre of Pressure Travel.

If the position of the centre of pressure of a flat plate, in terms of the chord, is plotted against the angle of incidence in degrees, the result is shown by the curve of Fig. 26. A C_P curve for a typical aerofoil is also shown on the same diagram.

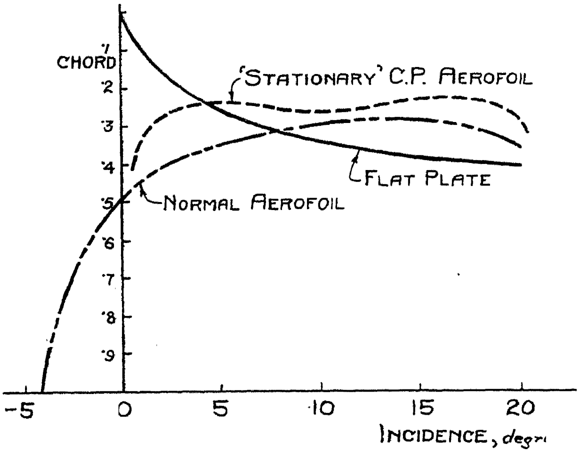


FIG. 26.—CENTRE OF PRESSURE TRAVEL

For the flat plate the C_P moves back from the quarter-chord position at a very small incidence, to about 0·4-chord position at 20°, finally arriving at the mid-position at 90°, whereas in the case of the aerofoil the C_P travels back from 0·3-chord at about 16°, and arrives at the trailing-edge at a small negative angle. Thus it is seen that the movements are in opposite senses.

Consider an aerofoil during flight at an angle α (Fig. 27 (*a*)), such that the weight W and air reaction R act at the same point and thus provide equilibrium. If the angle is increased to α_1, then R moves forward and a couple is set up tending to increase the disturbance. For the flat plate, however, when the incidence is increased from α to α_1 (Fig. 27 (*c*) and (*d*)), it is seen that the couple formed tends to restore the plate to the original position. For this reason the flat plate C_P movement is said to be stable, whilst the aerofoil C_P is referred to as having an unstable movement.

The size of the tail-plane of an aeroplane is dictated largely by the unstable C_P movement of the wings, from which it is realised that a cambered plane is disadvantageous in this respect.

Aerofoils of symmetrical section, and certain others with reflex curvature towards the trailing-edge, show little movement of the C_P, but such sections generally possess lift characteristics below the best.

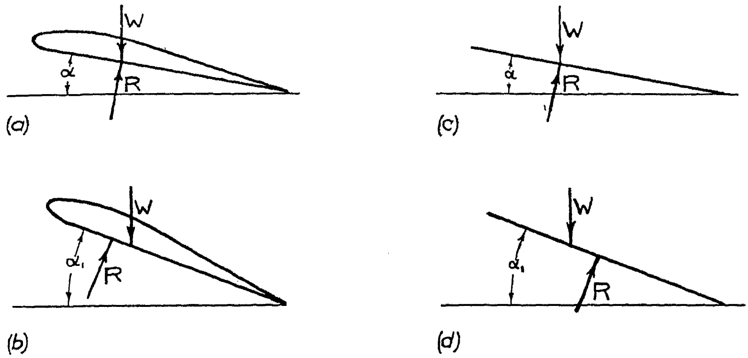


FIG. 27

CHAPTER IV

THE PROPERTIES OF AEROFOILS

Lift and Drag.

The aerofoil, or wing, is undoubtedly the most important part of the structure of an aeroplane. Practically the whole of the weight of the aircraft is supported in flight by the wing, whilst the wing weight may constitute nearly one-half of the structure weight and the wing structure account for roughly one-third of the drag at top speed. At lower speeds the proportion of wing drag increases considerably, and may form by far the greater part of the total drag at the minimum speed of flight.

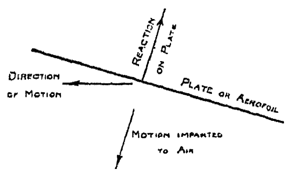


FIG. 28

Much depends on the designer's choice of a wing section, and his endeavour should be to obtain (*a*) an aerofoil that provides the necessary lift with small drag values over a large range of incidence, (*b*) a section that allows a good depth of spar compatible with the wing loading employed and length of wing between supports, and (*c*) the best disposition of the wing surface and arrangement of main planes in order to retain, with the minimum loss, the aerodynamic characteristics of the section employed.

When a flat plate, or aerofoil, is moved at some angle through the relative wind the air is given momentum in a downward and forward direction, whilst a reaction is induced on the plate (see Fig. 28).

It has been found that over the velocity range of use in present-day flight the reaction on the plate is proportional to the kinetic

energy of the stream of air influenced by the presence of the plate, which in turn is dependent on the plate area.

Hence we may write $R \propto \frac{1}{2} \frac{w}{g} V^2 S,$

where w = weight of air per cu. ft. in lb. (weight density),

or $R = C \frac{1}{2} \frac{w}{g} V^2 S$, where C is a coefficient, the value of which can be determined by experiment for each angle of incidence.

It is usual to write ρ for $\frac{w}{g}$, where ρ is called the mass density of air, its value under normal conditions at sea level being found as follows :—

$$\rho = \frac{w}{g} = \frac{0{\cdot}077 \text{ lb. wt. per cu. ft.}}{32{\cdot}2 \text{ ft. per sec. per sec.}}$$
$$= 0{\cdot}002378 \text{ slugs* per cu. ft.}$$

The formula therefore becomes

$$R = C \frac{\rho}{2} V^2 S,$$

and this, together with the value of ρ at sea level, should be remembered since the formula is widely used in aeronautics.

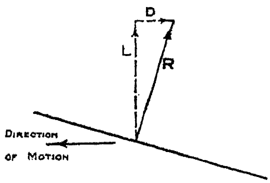


FIG. 29

The force R acts approximately at right angles to the plate, or to the chord line in the case of an aerofoil, and is divided for convenience into components parallel and perpendicular to the direction of motion (see Fig. 29). These components of the total force are called drag, D, and lift, L, respectively and may therefore be written

* The slug as a unit of mass is introduced thus. The *poundal* is that force which gives an acceleration of 1 ft. per sec.^{2} to a 1 lb. mass. This unit is considered too small for engineers, who have adopted the 1 lb. wt. (or simply 1 lb.) as the unit of force. This unit will give to 32·2 lb. an acceleration of 1 ft. per sec.^{2} A mass of 32·2 lb. (i.e. g lb.) is known as 1 slug.

$$D = C_D \frac{\rho}{2} V^2 S, \text{ and}$$

$$L = C_L \frac{\rho}{2} V^2 S$$

where C_D and C_L are the drag and lift components of the total coefficient of resistance, C, in the directions as specified above.

It will be realised that the lift, L, is the force supporting an aeroplane during flight, and D, the drag, is the resistance to forward motion that has to be counteracted by the engine through the propeller.

The values of C_L and C_D are found for all angles by wind tunnel tests.

As an example on the above, the main plane area required for a machine weighing 2,000 lb. with a minimum speed of 40 miles per hour, the maximum lift coefficient being 1·2, may be found as follows :—

The lift must equal the weight,

$$\text{Area} = \frac{\text{Weight}}{C_L \frac{\rho}{2} V^2} = \frac{2{,}000 \times 2}{1{\cdot}2 \times 0{\cdot}002378 \times (1{\cdot}466^* \times 40)^2}$$

$$= 410 \text{ sq. ft.}$$

Or again the velocity in m.p.h. necessary to maintain horizontal flight when the lift coefficient is 0·6, the weight and air density being unaltered, and the wing area as found above, will be :

$$(V \times 1{\cdot}466)^2 = \frac{2{,}000}{0{\cdot}6 \times \frac{\rho}{2} \times 410} = 688$$

$$V^2 = \frac{688}{(1{\cdot}466)^2} = 320$$

$$\text{and } V = 56{\cdot}7 \text{ miles per hour.}$$

This could have been done more directly by reasoning that since the lift coefficient has been reduced to one-half of the former value the square of the velocity must be doubled to compensate. Thus the new velocity would be $V_2 = \sqrt{2}\, V_1$

$$= 1{\cdot}414 \times 40$$

$$= 56{\cdot}7 \text{ m.p.h.}$$

* Ft. per sec. = 1·466 m.p.h.

Fig. 30 gives, in the usual form, the aerodynamic characteristics of the well-known aerofoil R.A.F.15.

For most aerofoils the *lift coefficient*, C_L, curve is almost a straight line from the angle of no lift to within 4°, or so, of the angle for maximum lift, that is the increase of lift with increase of incidence is practically constant over the range of angles used in flight. The increase of C_L averages about 0·1 per degree for a wing of infinite aspect ratio, the amount depending largely on the radius of curvature of the nose, being greater with a sharp leading edge, and less when the nose is blunt. The maximum value, $C_{L\ max}$ is generally found at about 16° incidence, beyond which point the curve descends, in some cases abruptly, but in others the falling off of lift is gradual.

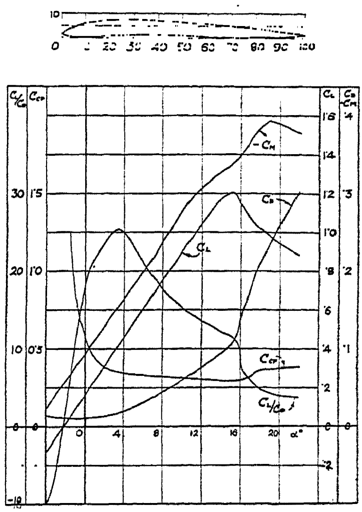


FIG. 30.—AERODYNAMIC CHARACTERISTICS OF AEROFOIL R.A.F 15

High values of $C_{L\ max}$ are most desirable for reducing the amount of wing surface required, or, alternatively, for the attainment of low landing speeds and good take-off qualities. These facts are obvious from the formula enunciated above,

$$L = C_L \frac{\rho}{2} V^2 S.$$

Low values of the *drag coefficient*, C_D, are consistent with low-powered engines, since the engine power is proportional to the drag of the aircraft, or again a low wing drag enables high maximum speed to be obtained. For top speed flight the wing incidence is generally arranged to be at, or near, the position of $C_{D\ min}$, and therefore a low value is particularly to be desired.

A figure of merit that combines the two above-mentioned qualities, and which therefore gives a rapid indication of the all-round performance of an aerofoil, is $\frac{C_{L\ max}}{C_{D\ min}}$ This is often used as a criterion for the comparison of different wing sections.

A rough measure of an aerofoil's efficiency is obtained from

the value of $\frac{L}{D}$ (max) since it determines the amount of load that can be carried per unit of propeller thrust, or, in other words, on it depends the gliding angle, which is a criterion of efficiency for aircraft.

The most efficient flying position for an aerofoil is at the incidence corresponding to $\frac{C_L}{C_D}$ (max) and for economy in flight this is roughly the position at which the wings should be set for cruising speed, some modification being made to allow for the parasite drag of fuselage and other parts.

The power required for flight is proportional to the product of drag and velocity. The former for an aeroplane of fixed weight and wing area is inversely proportional to $\frac{C_L}{C_D}$, whilst the latter is inversely proportional to $\sqrt{C_L}$, from which it is seen that power is inversely proportional to This quantity is known as the power factor, a high maximum value of which is consistent with small engine power for any given design, and also with good climbing qualities.

Centre of Pressure Travel, Moment Coefficient and Aerodynamic Centre.

The centre of pressure (C_P) is the point on the chord line at which the total air reaction would act if the distributed pressure were replaced by a single force R (see Fig. 31). The ratio $\frac{d_1}{c}$ is often called the C_P coefficient, d_1 being the distance of C_P from the leading-edge.

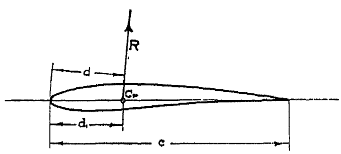


FIG. 31

The moment of the force about the leading-edge,

$$M = R\,d \qquad (6)$$

Also since $R \times d$ is very nearly equal to the lift component of R, multiplied by d_1, the moment equation can be expressed as

$$M \triangleq L\, d_1 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (7)$$

In order to express the equation in coefficient form, similar to the lift and drag coefficients, it may be written

$$M = C_M \frac{\rho}{2} S V^2 c \quad . \quad . \quad . \quad . \quad . \quad . \quad (8)$$

At ordinary angles of flight the moment is seen to give a turning effect about the leading-edge, by which the nose is depressed, and for this reason it is usual to show the moment coefficient curves as having negative values.

$$\text{From (7) } d_1 \triangleq \frac{M}{L} \triangleq \frac{C_M \frac{\rho}{2} S V^2 c}{C_L \frac{\rho}{2} S V^2} \triangleq \frac{C_M c}{C_L}$$

or the moment coefficient

$$C_M \triangleq \frac{C_L\, d_1}{c} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (9)$$

In the consideration of the tail-plane area the moments of the wing force about the leading-edge are translated into moments about the C_G of the aircraft, whilst, for the torsional strength of the main-plane spar, or spars, the moment about the spar location must obviously be considered. But in the early stages of a design, before spar and C_G positions are known, some other point of reference is necessary and hence the leading-edge has been chosen as datum.

With most aerofoil sections the centre of pressure position varies with the incidence, the most forward position, known as C.P.F., being about one-third of the chord back from the leading-edge, corresponding to $C_{L\ max}$ and to the largest incidence of general use in flight. Centre of pressure back, C.P.B., the position for fast horizontal flight, is at roughly $\frac{1}{2}$-chord, whilst for fast dives the C_P moves back toward, or even beyond, the trailing-edge according to the wing sectional shape. Thus the moment of the lift force must also vary, depending, to a large extent at least, on the C_P movement.

The disadvantages of a movable C_P may be summed up as (*a*) stronger wing structure necessary to allow for different loading dispositions ; (*b*) larger tail surfaces, or more accurately, tail " volume " (tail area $\times$ distance from the centre of gravity of aircraft) required ; (*c*) stronger fuselage to withstand heavier loadings of tail plane, and (*d*) heavier structures generally due to (*a*), (*b*) and (*c*).

From the above it is at once realised that a restricted range of movement of the C_P is a desirable feature for an aerofoil, although unfortunately it is difficult of attainment, and considerable research has been undertaken with the object of producing sections incorporating this quality.

There are two well-known methods of obtaining a stationary, or nearly stationary, C_P. The first is to employ a symmetrical section, such as R.A.F.30, whilst the second is to give the aerofoil a reflex curvature towards the rear, i.e., to turn the trailing-edge slightly upwards.

The disadvantages of these methods are that the values of $C_{L\ max}$ are not so good as with certain other sections, whilst there is an increase in drag with reflexed aerofoils.

The desired effect has been more recently achieved by designing the mean-line shape so as to give decreasing curvature from the leading-edge aft and arranging for the rear part of the mean line to be devoid of curvature. Tests* made with such aerofoils have given very satisfactory results and will be dealt with later.

Aerodynamic Centre.

The pitching moment of an aerofoil remains virtually constant throughout the flight range when considered about a point situated on the chord line at approximately one-quarter of the chord back from the leading-edge, and for this reason the moment may be given the one value instead of being plotted as a curve with the leading-edge as the point of reference. More recently the exact location of the point about which the pitching moment is constant has come into use, and this point is termed the Aerodynamic Centre.

The location of the Aerodynamic Centre relative to the $\frac{1}{4}$-chord position is then given by horizontal and vertical offsets in terms of x/c and y/c.

* *N.A.C.A. Reports*, Nos. 530 and 537.

Structural Considerations.

An aerofoil possessing good values of the factors so far considered would be of little use if its shape did not allow for the proper housing of spars. For instance, deep spars cannot be accommodated in shallow wing sections, and the extra weight involved in obtaining the requisite strength in such a case may more than outweigh the benefit of good aerodynamic qualities. In the past the most favoured method of wing construction, in this country at least, has been the two-spar system, for which the depth available at the rear spar in many aerofoils is rather small. The present-day tendency to increase the span of bays, or distance between supports, has brought about the desirability for deeper wing sections, whilst the problem of obtaining sufficient depth for the rear spar has become more acute.

This difficulty is being overcome by some designers by the employment of a single spar, which is situated at or near the position of maximum depth, so that the use of more efficient aerofoil sections has become possible.

In summation, from the foregoing it is seen that the desirable features of a wing section are :

(*a*) High value of $C_{L\ max}$.
(*b*) Low values of C_D and particularly of $C_{D\ min}$.
(*c*) High $\dfrac{C_{L\ max}}{C_{D\ min}}$.
(*d*) High value of $\dfrac{C_L}{C_D}$ (max).
(*e*) High $\dfrac{C_L^{1.5}}{C_D}$ (max).
(*f*) Small range of centre of pressure travel, and
(*g*) Suitable depth for the accommodation of spars.

Polar Curve for Aerofoil.

A method of showing graphically the characteristics of aerofoils, considerably used on the Continent, and to an increasing extent in this country, is known as the polar curve, shown in Fig. 32, in which C_L is plotted against C_D. The angles of incidence, for which the points are plotted, are marked on the polar.

Values of $\dfrac{C_L}{C_D}$ can quickly be obtained by dividing the vertical

ordinate by the abscissa value, whilst $\frac{C_L}{C_D}$ (max) is found by drawing a line through the origin, tangential to the polar.

For example, for the wing section R.A.F.34 of Fig. 32 the tangent has been drawn and touches the polar at approximately the 4° position, for which C_L and C_D are 0·4 and 0·02 respectively, so that $\frac{C_L}{}$ (max) $= \frac{0{\cdot}4}{0{\cdot}02} = 20$.

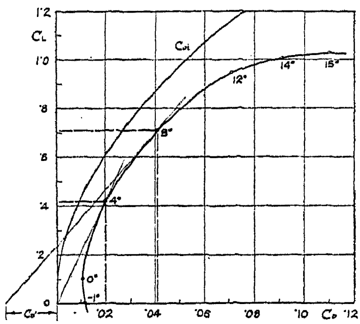


FIG. 32.—POLAR CURVE FOR AEROFOIL R.A.F. 34 (ASPECT RATIO 6)

Comparisons between aerofoils can be more quickly made from polar diagrams than from separate curves for lift, drag, and lift /drag.

It has been seen that the most efficient position for the wing alone, for the aerofoil under consideration, is at an angle of 4° for which $\frac{L}{D}$ is 20. If the drag of fuselage, landing gear, tail unit, etc., are to be taken into account there is an increment of drag to be added to the wing drag at all angles.

The increase to the drag coefficient, which is termed the parasite, or body drag, and will be more fully dealt with later, can be obtained by dividing the drag of all parts, except the wings, by $\frac{\rho}{2} S\ V^2$

$$\text{or } C_{D^1} = \frac{D^1}{\frac{\rho}{2} S\ V^2} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (10)$$

For example, if the parasite drag of a small monoplane of 200 sq. ft. area is given as 47·4 lb. at 100 ft. per sec., then

$$C_{D^1} = \frac{47{\cdot}4}{\frac{{\cdot}002378}{2} \times 200 \times (100)^2} = 0{\cdot}02$$

Now this increment to the drag coefficient could be added to the values of the wing drag coefficient to obtain the drag

coefficient for the complete machine, which will result in an alteration of the $\frac{L}{D}$ values.

However, where the polar diagram is employed the result can be simply achieved by setting off the parasite drag coefficient to the left of the origin, so that all drag values may now be read from the new origin (see Fig. 32).

In this way the value of $\frac{L}{D}$ (max) for the whole aeroplane can be quickly found by drawing a straight line from the new origin to touch the polar as before. In this case the polar is touched at roughly 8° and the $\frac{L}{D}$ ratio is $\frac{0{\cdot}710}{{\cdot}042 + {\cdot}02} = 11{\cdot}44$, say.

This determines the best gliding angle that could be obtained with the combination of wing and body as outlined, and at the same time indicates the angle of incidence at which the wings should be set to obtain this result.

Induced Drag.

Consider first a rectangular wing of finite span but fitted with end plates, or walls, to produce two-dimensional flow. The lift will be uniform across the span, and the air-flow on passing the wing will be given a downward velocity, which also will be constant across the span. (Fig. 34 (*a*).)

Now imagine that the end plates are removed. At the wing tips there will be a tendency for the air to flow from the underside to the topside, from the region of increased pressure to one of reduced pressure. Above the wing there will be a spanwise inflow, and below the wing an outflow will take place. (Fig. 33 (*a*).) These flow components combine to produce a rotary flow, shown at (*b*), and result in the formation of a vortex, having its core, or axis, approximately along the wing-tip chord and its source just forward of the leading-edge (see Fig. 33 (*c*)).

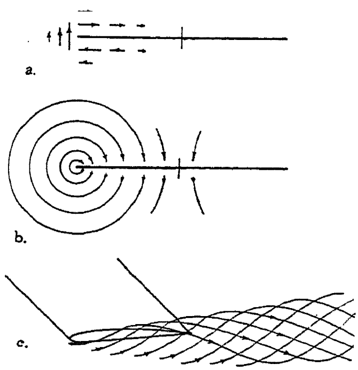
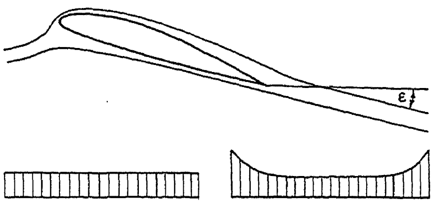


FIG. 33.—WING-TIP VORTEX

The existence of the pair of wing-tip vortices causes an additional downwash over the wing, the effect being most pronounced at the tips. (Fig. **33** (*b*).) It might at first be thought that this extra downwash would result in an increased lift, but in fact there is a falling off in lift towards the wing-tips due to the neutralisation of pressures there. Viewing the system as a whole it will be seen that there is no nett increase in downwash, since there is an accompanying upwash beyond the tips.

It may be mentioned in passing, however, that at large angles of incidence, beyond the normal stalling angle, the lift distribution across the span may show greater intensity towards the wing-tips. This is due to the delayed stalling over the outer portions of the wing, where the vortex is most active, and provides one aspect of the vortex effect that is favourable in practical flight.



(*a*) *Two-Dimensional Flow* (*b*) *Three-Dimensional Flow*

FIG. **34**.—DOWNWASH ACROSS SPAN OF RECTANGULAR WING

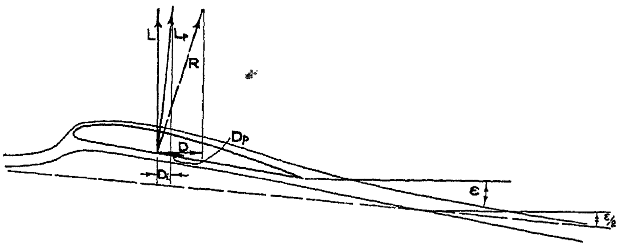


FIG. 35.—INDUCED DRAG

The reaction, R of Fig. **35**, is normally divided into components perpendicular and parallel to the line of flight, denoted by L and D respectively, since these are the force components that balance out in flight with weight and thrust.

It would be more scientifically correct to split R into components based on the mean relative flow, which is inclined downward from the horizontal at an angle $\frac{\epsilon}{2}$, where ϵ is the downwash

angle. These components give the true lift and drag, called profile lift and drag, and denoted in Fig. 35 by L_p and D_p.

L_p is substantially equal to L, but the same cannot be said of D_p and D. It will be seen that the resistance of the wing to forward motion consists of a component of D_p (roughly equal to D_p) plus a component of the true lift, called the induced drag, $D_i\left(= L_p \sin \frac{\epsilon}{2}\right)$.

The profile drag consists of form drag and skin friction and is dependent on the aerofoil shape and its attitude to the mean relative wind, but is not affected by aspect ratio, or plan form, whilst the induced drag depends entirely on lift and downwash, and has a constant value for all wing-sections, having the same plan form, for equal values of lift.

Notice that for a finite aspect ratio, if no downwash is present, lift is zero and the only force acting is profile drag. Similarly for an infinite aspect ratio, downwash is infinitely small so that L_p coincides with L, and again there is no induced drag. In other words the induced drag for a given lift depends on the aspect ratio of the aerofoil, and its value may be determined if ϵ can be found.

The wing lift reaction must be equal to the downwash momentum imparted to the air, or to the mass of air acted upon in unit time, multiplied by the downward velocity. If S′ equals the "swept" area, or area affected, then

$$L = S' \rho V \times V \sin \epsilon, \text{ or since } \epsilon \text{ is small}$$

$$L = S' \rho V^2 \epsilon, \text{ where } \epsilon \text{ is in radians} \quad . \quad . \quad . \quad . \quad . \quad (11)$$

The mean wind, or induced, angle is then

$$\frac{\epsilon}{2} = \frac{L}{2 S' \rho V^2} \qquad (12)$$

Now, by Prandtl, the total swept area is approximately *equivalent* to a circle of diameter equal to the span, or $S' = \frac{\pi b^2}{4}$.

The induced drag therefore is the product of the lift (profile) and the induced angle (in radians), or

$$D_i = \frac{L^2}{2S' \rho V^2}$$

$$= \frac{L^2}{\frac{\pi}{2} b^2 \rho V^2} \quad \ldots\ldots \quad (13)$$

And the coefficient of induced drag

$$C_{Di} = \frac{D_i}{\frac{\rho}{2} S V^2}$$

$$= \frac{L^2}{\frac{\pi}{4} b^2 \rho^2 S V^4}$$

$$= \frac{C_L{}^2 \frac{\rho^2}{4} S^2 V^4}{\frac{\pi}{4} b^2 \rho^2 S V^4}$$

$$= \frac{C_L{}^2 S}{\pi b^2}, \text{ or since } \frac{b^2}{S} \text{ is the aspect ratio, A,}$$

$$C_{Di} = \frac{C_L{}^2}{\pi A} \quad \ldots\ldots \quad (14)$$

Values of the induced drag coefficient C_{Di} for the most commonly used aspect ratios are given in Table II below. The polar curve for one aspect ratio is also plotted in Fig. 32.

TABLE II.—INDUCED DRAG COEFFICIENTS

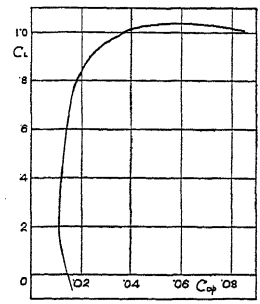
C_L	Aspect Ratio				
	5	6	7	8	9
0·1	0·0006	0·0005	0·00046	0·0004	0·00035
0·2	0·0025	0·0021	0·0018	0·0016	0·0014
0·3	0·0057	0·0048	0·0041	0·0036	0·0032
0·4	0·0102	0·0085	0·0073	0·0064	0·0057
0·5	0·0159	0·0133	0·0114	0·0100	0·0087
0·6	0·0229	0·0191	0·0164	0·0143	0·0128
0·7	0·0312	0·0260	0·0223	0·0195	0·0173
0·8	0·0408	0·0340	0·0292	0·0255	0·0227
0·9	0·0515	0·0430	0·0369	0·0322	0·0286
1·0	0·0637	0·0530	0·0455	0·0398	0·0354
1·2	0·0918	0·0764	0·0657	0·0573	0·0510
1·3	0·1077	0·0898	0·0771	0·0673	0·0598
1·4	0·1248	0·1040	0·0894	0·0780	0·0694

Although not strictly correct, especially at large angles of incidence, the profile drag is assumed to be the difference between the total and calculated induced drags.

Since the induced drag is practically the same for all aerofoils at the same aspect ratio, it is quite usual to include the curve for C_{Di} (a parabola) with the polar curve of each aerofoil. This has been done in Fig. 32 and gives C_{Di} for an aspect ratio of six.

In this way the profile drag can be quickly read off for any lift value and is obtained by deducting the abscissa for C_{Di} from the abscissa for total drag.

The profile drag for each wing section can be plotted in polar form. Fig. 36 gives the profile drag plotted against C_L for R.A.F.15.



.—PROFILE
R.A.F.

Angle of Attack and Aspect Ratio.

An increase of aspect ratio is accompanied by reduced drag values. Due also to the reduction of downwash the true effective angle of attack, $\alpha - \frac{\epsilon}{2}$, is increased, or for a specific lift value the apparent, or geometric, angle is reduced. The reduction in the angle of attack is equal to the difference between the induced angles for the two aspect ratios.

The induced angle $\frac{\epsilon}{2} = \frac{L}{2\ S'\ \rho\ V^2}$,

or substituting again for L and S′

$$\frac{\epsilon}{2} = \frac{C_L\ \frac{\rho}{2}\ S\ V^2}{\frac{\pi}{2}\ b^2\ \rho\ V^2}$$

$$= \frac{C_L\ S}{\pi\ b^2} \text{ or } \frac{C_L}{\pi A}$$

Reverting to degrees instead of radians :

$$\frac{\epsilon}{2} = \frac{18{\cdot}25\ C_L}{A} \qquad . \qquad . \quad . \quad (15)$$

The change in the angle of attack for an alteration of aspect ratio, in order to retain the same lift value, is then

$$\alpha_2 = \alpha_1 - \left(\frac{\epsilon_1}{2} - \frac{\epsilon_2}{2}\right) \quad \text{or}$$

$$\alpha_1 - \alpha_2 = 18{\cdot}25\ C_L \left(\frac{1}{A_1} - \frac{1}{A_2}\right) \quad . \quad . \quad (16)$$

The suffixes 1 and 2 relate to the smaller and larger aspect ratios respectively.

Angle of Downwash.

The average downwash angle immediately behind an aerofoil is seen above to be twice the value given in (15), or

$$\epsilon = \frac{36{\cdot}5\ C_L}{A} \quad . \quad . \quad . \quad . \quad . \quad . \quad (17)$$

CHAPTER V

VARIATIONS OF AEROFOIL SHAPE AND THEIR EFFECTS ON AERODYNAMIC CHARACTERISTICS

GENERALLY speaking, it is doubtful whether the designer should attempt to modify existing aerofoil sections unless a wind tunnel is available to him, and even then modifications are best confined to those dictated by structural requirements unless extensive research is to be carried out.

However, a good general knowledge of the effects of variation of such qualities as degree of camber and thickness ratio should prove of undoubted assistance in the selection of a section for a specific design, and for this reason alone it is well that the effects of variations of thickness, camber, etc., should be understood.

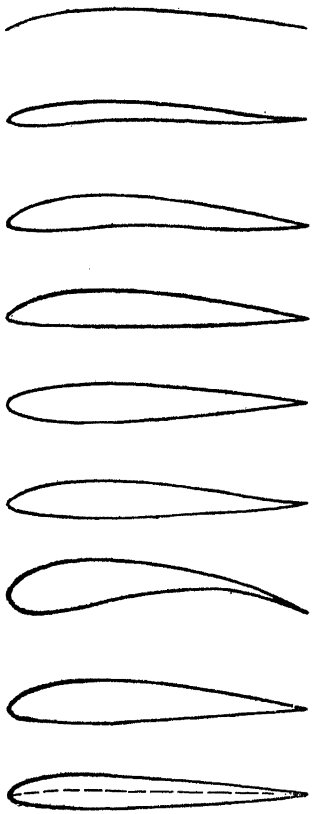
Evolution.

The unsuitability of a flat plate as a lifting surface has already been briefly touched upon. Early experimenters appreciated the desirability of a cambered surface, and many of the early flight attempts were made with what was little more than a cambered plate having inappreciable depth. This was followed by double surface aerofoils, of very elementary and arbitrary shape, with but little importance attached to thickness of the section, the wing strength being obtained generally from a network of external bracing wires.

In 1915, one of the best-known aerofoils, the R.A.F.15, was introduced. This aerofoil section, although thin as compared with present-day standards, was deep enough to house front and rear spars of sufficient depth for moderately loaded biplanes, gave low drag values over a good range of incidence, and possessed good general aerodynamic qualities. This section was probably used in a greater number of designs than any other aerofoil, and certainly it had a longer life, having been freely employed over a period of at least twelve years. The main factor preventing the R.A.F.15 from a place in present-day aircraft is

its limited depth, which prohibits the use of adequate spar sizes and the employment of split flaps.

Following the R.A.F.15 aerofoil came a later member of the series, the R.A.F.19, in which a very deep camber was the outstanding feature. This was known as a "high-lift" section, its value for $C_{L\ max}$ being in the neighbourhood of 1·8, but the accompanying drag values prevented the attainment of rapid flight, $C_{D\ min}$ being 0·07, and $\frac{L}{D}$ (max) being no greater than **11·8**.



SECTION	CAMBER	THICKNESS
Elementary Aerofoil		
R.A.F. 15	2·5	6·5
U.S.A. 27	5·83	11·9
Clark YH	3·04	11·7
R.A.F. 30	0·0	12·64
R.A.F. 34	4·16	12·64
Gött. 652	7·0	19·2
R.A.F. 48	3·0	15·0
N.A.C.A. 23012	1·85	12·0

FIG. 37.—HISTORICAL DEVELOPMENT OF AEROFOILS

The need for spars capable of resisting the higher bending moments due to greater spacing of supports, heavier wing loadings, and the monoplane structure, led to the introduction of medium

thickness aerofoils having a depth of about 12 per cent. of the chord. Their advent had been delayed on account of the unfavourable scale effect of such sections when tested at the slow wind-tunnel speeds then available. In Germany a series of Göttingen sections was produced, their distinguishing feature being a forward position of the point of maximum camber, with a well-rounded leading-edge.

The U.S.A.27, developed in 1919, was obtained by doubling the ordinates of the successful R.A.F.15.

These were followed by the Clark Y (1922), another most successful section and one that was widely used over a period of several years; a development, the Clark YH, being in use to-day, together with a series of R.A.F. sections, Nos. 30 to 33 (1924), in which the reduction of C_P movement had been sought and largely obtained. By curtailing the C_P travel, lighter wing structures and smaller tail-planes became possible. The basic section of the series, R.A.F.30, was symmetrical about the chord line, and gave a $C_{D\ min}$ of only 0·01, whilst the others were developed from this first by cambering the median line for greater lift, and then by reflexing the trailing-edge to restore the constant C_P location.

In Germany a series of "water drop" sections was developed, having considerable depth over the front portion, but tapering off very markedly towards the rear, the lower surface having a convex curvature forward changing to one of considerable concavity behind. These sections gave high values of $C_{L\ max}$, these being 1·56 for Göttingen 535 (1926), and 1·8 in the case of Göttingen 652 (1928), but with high drag, and $\frac{L}{D}$ (max) of 17 only. The centre of pressure travel was also large with correspondingly high pitching moments. The high lifts of these sections made them very popular for sailplane work, whilst their shapes lent themselves to the single-spar type of wing, with stiff leading-edge torsion tube, favoured in the construction of gliders.

The desire for still greater spar depth led to the introduction of the R.A.F.48 (1929), with a thickness of 0·15 c, and even deeper sections. The depth and camber of R.A.F.48 provide good high-lift values, whilst the bi-convex shape ensures relatively low drag at small angles of incidence, so that this is regarded as one of the best all-round sections to-day.

Different desiderata were responsible for the development of the N.A.C.A.23012 aerofoil (1935), the characteristics of which compare most favourably with other present-day sections. Earlier investigation had indicated that higher values of $C_{L\ max}$ were obtainable with sections having a forward position of the point of maximum camber.* The section considered was given a small amount of camber only for low drag, but with its maximum located at 0·15 c from the leading-edge for relatively high lift. The mean-line curvature decreases steadily from the leading-edge and in fact the rear three-quarters of the mean line is straight, a feature which helps not only to preserve low drag but also gives low pitching moments. The N.A.C.A.23012 suffers from one disadvantage, the sharp break in the C_L curve at the incidence of maximum lift.

The characteristics of several of these aerofoil sections are given in the Appendix† and on pages 43 and 48.

The drag of good, thick sections is but little inferior to the earlier thinner sections, and is more than compensated for by the saving of resistance due to the internal accommodation of all, or most, of the structural members.

Mean-Line Camber.

The camber of the median line of an aerofoil section, its degree and distribution, have come to be looked upon as the most important features, since it is upon camber that so much of the aerodynamic qualities depend. Thickness and camber are closely bound together and the effects of varying one can often be largely reversed by discretionary variation of the other.

The effect of camber as shown by the lift curve is a uniform increase of lift values by moving the curve and no-lift angle one degree of incidence to the left for each per cent. increase of camber. The increase of $C_{L\ max}$ is not so pronounced as the lift increase at the smaller angles of incidence, whilst the incidence at which maximum lift occurs is practically independent of camber. There is one other important point in connection with the lift curve and this concerns the peak, which becomes progressively more rounded as the camber increases.

This applies chiefly to sections of moderate thickness, in the neighbourhood of 12 per cent., which are well-known to stall

* See p. 46. † Volume II.

sharply, and the remedy is seen to be an increase of centre line camber.

The increase of $C_{L\,max}$ with camber is roughly 5 or 6 per cent. for each one per cent. camber increase in the case of thin sections, the effect becoming less pronounced with thick sections, and with sections having a forward position of maximum camber.

The angle of no-lift for a symmetrical section is of course zero, and, from what has been said above, it is seen that the angle of incidence for no-lift for any section is roughly equal in degrees to the camber /chord percentage.

Minimum profile drag increases but slowly with camber up to 6 per cent., but above this the drag increase becomes progressively greater, and a 6 per cent. mean-line camber is seldom exceeded for this reason.

If $C_{Dp\,min}$ were the only consideration a very small camber would obviously be adopted, but the profile drag co-efficients corresponding to other lift values are also of importance, depending on the use to which the aircraft is to be put, and the two qualities must be considered together. In other words there is a best camber for each value of C_L. Where top speed is the criterion a camber of about one per cent. would be most suitable and would show a smaller drag than a symmetrical section. If the climbing attitude is of importance a camber from 2 to 4 per cent. should be selected.

Camber is also, as might be anticipated, the determining factor as regards pitching moment, the value of the moment coefficient about the quarter chord at the no-lift angle being approximately $-0{\cdot}02$ for each per cent. of camber, decreasing however, for a forward location of the maximum camber, but increasing as it is moved towards the trailing-edge.

In general, an aerofoil with a camber between 2 and 4 per cent. should be chosen for good all-round performance, the preference being with the smaller value, and it may be mentioned that the selection as indicated above is of greater importance with thin, than with medium and thick sections.

Position of Maximum Camber.

So far little has been said of variation of the location of the point on the chord line of the maximum camber, though it is not of great importance. A forward position gives greatest lift for a thin section, though drag also increases, whilst for

thick sections exactly the reverse holds good. Reference has already been made to the beneficial pitching-moment effect obtained from a forward position of the point of maximum camber, and this is perhaps the most important feature, since tail volume and structural weight depend largely on this quality. On the whole, therefore, a location at 0·3 *c*, or further forward, appears desirable.

The results of recent tests in America* on a family of aerofoils with the position of maximum camber unusually far forward showed such sections to possess exceptionally low pitching moments.

Camber Distribution.

It is only recently that systematic research† has been undertaken to ascertain the effects of change of distribution of camber without alteration of the maximum amount. It was then found that by moving the point of maximum camber of the mean line of a section either more forward or more rearward higher values of $C_{L\,max}$ resulted. Pitching moments on the other hand became greater as the maximum camber moved back, and improved for the forward locations, and on this account no further tests were made with the former arrangement.

$C_{L\,max}$ continues to increase slightly up to the most forward position tested, 0·05 *c*, and this is accompanied by slight drag decrease, and, as mentioned above, decreased pitching moment with a zero value about the quarter-chord location for a maximum camber position six per cent. from the leading-edge. A feature of importance seems to be the need for the mean line to have its maximum curvature right forward, with a gradual decrease rearward, and with no curvature at all over the rear two-thirds of the chord.

The tests described above were not sufficiently comprehensive to allow the results to be assumed valid for all basic aerofoil shapes, but they are probably correct for sections of some 12 per cent. thickness with moderate camber.

Aerofoil Thickness.

Within certain limits the effect of increasing the depth of an aerofoil is to augment the lift values and $C_{L\,max}$. This

* *N.A.C.A. Report*, No. 537. † *N.A.C.A. Reports*, Nos. 530 and 537.

is as would be expected, since it is equivalent to an increase of the top surface camber. The effect of thickening is greatest with aerofoils of symmetrical section and decreases as the mean-line camber becomes more pronounced. Thus the optimum thickness for $C_{L\ max}$ of a symmetrical aerofoil is about 13 per cent., 11·5 per cent. being the figure for a mean-line camber of 0·03 *c*, and 10 per cent. for a camber of 0·06 *c*. By moving back the position of maximum mean-line camber, a little further thickening becomes possible with increase in $C_{L\ max}$ due of course to the smoother top-surface curve and its lesser tendency to cause separation to take place.

The loss of maximum lift with greater thicknesses than the optimum is gradual, but is rapid as the depth is decreased.

The peak of the lift curve is well rounded for thin sections, becomes very sharp for a thickness ratio of about 12 (the thickness for which $C_{L\ max}$ is greatest), and returns to a smooth curve as the depth is still further increased. But it should be noted that the angle of $C_{L\ max}$, and hence the angular range, increases also with thickness up to 12 per cent., beyond which it retreats slightly. A thickness of 20 per cent. has not only a relatively high value of maximum lift, but also an almost imperceptible manifestation of stalled conditions. This suggests that the stall of an aircraft will be most catastrophic when the wing has a depth of 0·12 *c*, and furthermore that at the wing-tip the thickness should be less, or more, preferably the latter, than 12 per cent. to prevent the phenomenon known as "dropping a wing."*

The effect on drag of thickness is a uniform increase of from 4 to 5 per cent. of the minimum profile drag for each per cent. increase in depth; this being little more than one-third of the increase in $C_{L\ max}$ below the optimum thickness, so that the value of $\frac{C_{L\ max}}{C_{D\ min}}$ is greatest for a thickness giving the highest $C_{L\ max}$, or very slightly below. Recent experiments† on aerofoils of small camber and varying depths, of the N.A.C.A. 23012 series, showed the greatest value of $\frac{L}{D}$, approximately 25, for a thickness of 6 per cent., but the ratio dropped only to 20 with a thickening up to 18 per cent. $C_{L\ max}$ was greatest at 1·68 for the 12 per cent. depth.

* But see p. 60.

† *N.A.C.A. Tech. Note*, No. 567.

A good average thickness for present-day wings is 12 per cent. of the chord, and in the case of the tapered wing monoplane the thickness may vary from 0·15 *c* to 0·18 *c* at the root, or even 0·2 *c* if considered desirable for structural reasons, to between 0·07 *c* and 0·1 *c* at the wing-tip, the total drag of such a wing being little greater than that of the best thin section, excluding the drag of necessary supporting members in the latter.

Leading-Edge.

In general the blunter the leading-edge curve, the higher the value of $C_{L\ max}$. This is to be expected. But with the lift increase comes a sharper peak of the lift curve and a sudden drop of the curve after $C_{L\ max}$ is reached. There is a slight decrease of profile drag at the smaller angles of incidence with sharpening of the leading-edge curvature.

The leading-edge radius should be within 1 and 5 per cent. of the chord, but for purely racing purposes, if a high value of $C_{L\ max}$ is not considered necessary, or is to be obtained by means of some variable-lift device, a sharper leading-edge might be employed with advantage.

Reflexed Trailing-Edge.

Mention has been made of the method of restricting the C_P travel by giving a reflexed curvature to the rear portion of an aerofoil, the benefits accompanying such a limitation being obvious. Approximately the reflex, or vertical shift of the trailing-edge, measured in terms of the chord, counteracts the pitching moment brought about by an equal amount of mean-line camber. That is to say a section having a camber of 3 per cent. would give a moment coefficient of 0·06,* but by raising the trailing-edge a distance also 0·03 *c* the moment would become zero. The effect on the angle of zero lift is likewise numerically the reverse of that due to camber.

Tests in America† on three different aerofoils, both with and without reflex, showed results that agreed amongst themselves as well as with work carried out in this country at an earlier date. There is a loss of $C_{L\ max}$ of roughly 4 per cent. for each degree of reflex and a slight increase in drag values. Fig. 38 shows the curves for one pair of sections of the test referred

* See p. 59. † *N.A.C.A. Report*, No. 388.

to, for which the reflex angle was 3°, the scales relating to an infinite aspect ratio.

Still more recent wind tunnel work* on a family of related aerofoils, both with and without reflex, confirmed the adverse effect, as regards both maximum lift and minimum drag, previously noted. The same tests, the primary findings of which consisted in the discovery of aerofoil sections having a simple mean line and possessing very low pitching moments, appear to have shown that the reflexed aerofoil sections are surpassed by simple mean-line sections of good design and that the former may now be relegated to the past.

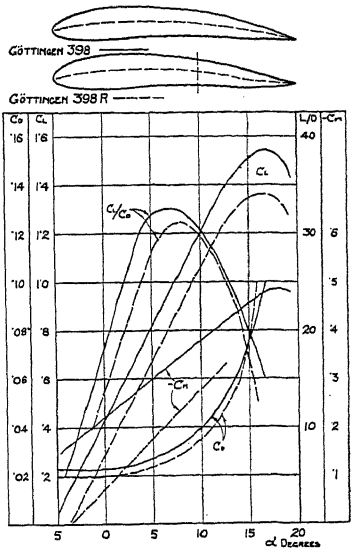


FIG. 38.—CHARACTERISTIC CURVES FOR GÖTTINGEN 398 (WITH AND WITHOUT REFLEXED TRAILING-EDGE)

Reynolds' Number.

Brief reference has already been made† to the effect of change of Reynolds' number on the aerodynamic characteristics of aerofoils. Certain tests have been carried out‡ with the main objective of ascertaining the effect of scale on the aerodynamic qualities of an aerofoil, and although they throw considerable light on the subject the results are not yet sufficient to be regarded as conclusive.

In general $C_{L\ max}$ increases with R.N. from values between 1·0 and 1·2 at 10^6, to between 1·4 and 1·5 at 6×10^6, though this does not hold good in all cases. At least two sections (Göttingen 387 and U.S.A. 35A), both thick sections of large mean camber and with relatively sharp trailing-edges, show a high maximum lift at low Reynolds' numbers ($C_{L\ max}$ 1·5 and 1·6

* *N.A.C.A. Report*, No. 537.

† See p. 8.

‡ "Tests of Six Aerofoil Sections at various Reynolds' Numbers in the Compressed Air Tunnel," *R. & M.* 1706. "Scale Effect on Clark Y Airfoil Characteristics from N.A.C.A. Full-scale Wind Tunnel Tests," *N.A.C.A. Report*, No. 502.

respectively, at R.N.10^6) but with values decreasing to 1·4 at 4×10^6.

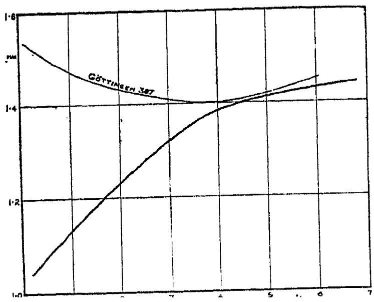


FIG. 39.—EFFECT OF REYNOLDS' NUMBER ON C_L MAX (AVERAGE OF 5 AEROFOILS)

A second generalisation is that the maximum lifts of all aerofoils tend to stabilise at a uniform value at high Reynolds' numbers, somewhere within the neighbourhood of 10^7, and the little evidence at present available appears to show a gradual falling off after the peak. Similarly the peaks of lift curves tend to sharpen with increase of R.N., this being a common characteristic of C_L curves where the maximum value is high.

Drag varies little with R.N. except below a value of 10^6 where C_D increases considerably. There appears to be an optimum value of R.N., roughly $2{\cdot}5 \times 10^6$, where $C_{D\,min}$ and minimum profile drag are least. Below this there is a fairly rapid increase, but the values are substantially constant above, with, if anything, a slight increase.

Minimum profile drags conform fairly closely to the turbulent skin-friction curve, assuming a plate area of 2 S to allow for the upper and lower surfaces. Sections of 12 per cent. thickness and less fall below the turbulent skin-friction curve at low Reynolds' numbers, but cross over at approximately R.N. of 3×10^6. Aerofoils having a thickness of $0{\cdot}15\,c$ lie close to the turbulent curve at very low R.N., but may be nearly 50 per cent. higher at 10^7.

Chapter VI

ARRANGEMENT OF LIFTING SURFACE AND TYPES OF AIRCRAFT

The Plan Form—Aspect Ratio.

The most important feature of an aeroplane wing is the proportion of the length, or span, to the chord, known as aspect ratio. Aspect ratio is defined as : $A = \frac{\text{span}}{\text{mean chord}}$.

With most aircraft some degree of taper, or rounding of the wing-tips, is employed so that the average chord is not immediately apparent. It can, however, be obtained by dividing the span into the area ; or average chord $= \frac{\text{area}}{\text{span}}$. Hence aspect ratio may be defined as $\frac{\text{span}^2}{\text{area}}$ and this is the form generally adopted.

The importance of aspect ratio has already been considered in Chapter IV under the heading " Induced Drag."* There is a slight increase in $C_{L\ max}$ with increase of aspect ratio, amounting to approximately 1·5 per cent. for each unit increase and in $\frac{C_L}{C_D}$ (max) of roughly 8 per cent. for each unit increase. The significance of this for aircraft likely to operate at the incidence appertaining to $\frac{C_L}{C_D}$ (max), air liners, sailplanes, etc., is readily apparent. For racing aircraft, on the other hand, where the qualities relating to the conditions obtaining at $C_{D\ min}$ are of chief importance, the advantage of high aspect ratio is less apparent, but it will be shown later† how aspect ratio affects take-off and climb, a high value being consistent with a good performance in both.

The choice of aspect ratio is dictated by two conflicting requirements, that of large span for aerodynamic efficiency, and

* See p. 51. † See Chapter XIII.

that of small span for structural efficiency, or lightness. It has been seen* that in order to keep profile drag within reasonable limits, the depth of a wing should not greatly exceed 0·15 c and 18 to 20 per cent. may be regarded as the maximum.

With this proviso, the weight of a wing of given area increases roughly as the square of the span. The rate of increase of aerodynamic efficiency, on the other hand, falls off with increasing aspect ratio and above 9 the gain is not great. Hence if aerodynamic efficiency alone were considered, an aspect ratio of 9 could be regarded as very satisfactory, but when structural weight is also taken into account a ratio of about 7 appears desirable.

Average figures for present-day utility aeroplanes vary between 6 and 9, the higher figure being used where good climing and economy of flight are of paramount importance, whilst smaller values are made use of for high speed aircraft. Aspect ratios of 20 are quite common for sailplanes, and as high a value as 26 has been employed. In this case aerodynamic efficiency is the main consideration, the additional wing weight being to some extent compensated for by extra wing area.

The aspect ratios of biplanes are given for each wing separately, since there are, of course, losses due to the end effect at the tips of both wings.

Wings of Unusually Small Aspect Ratio.

A small number of aeroplanes have been produced having a main plane of aspect ratio 1·0, or thereabouts. The plan form has taken the shape of a circle, square, triangle and other varied forms. In discussing aspect ratio effects it was stated that $C_{L\ max}$ increases with aspect ratio, and for normal wing forms this is true. Decrease of aspect ratio beyond a certain low value, in the neighbourhood of 1·5, introduces the curious feature of a rapid increase of maximum lift, a phenomenon first discovered with the flat plate in the earliest days of heavier-than-air flight.

The absolute maximum value of $C_{L\ max}$ is found at an aspect ratio of approximately unity, or rather less, below which there is again a rapid falling-off. $C_{L\ max}$ at the small aspect ratio indicated above may be as much as 20 per cent. greater than at the ratios normally used in flight.

* See p. 61.

The stall of such aerofoils, which takes place at a high angle of attack, roughly twice that appertaining to wings in common use, is sudden and is accompanied by an instantaneous loss of lift.

Tapered Wings.

It would appear that by tapering the main plane from root to tip, with consequent reduction of the wing-tip chord, there would be a diminution of the wing-tip losses and greater efficiency would result (see Fig. 40). This to some extent is true, but it

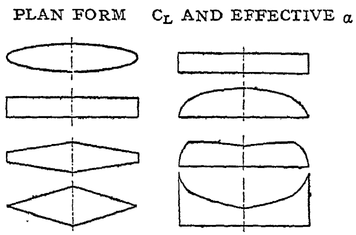


FIG. 40.—DISTRIBUTION ACROSS SPAN OF C_L AND EFFECTIVE α FOR RECTANGULAR AND TAPERED WINGS WITHOUT TWIST

has been found by experiment that a tip chord of from one-quarter to one-half the root chord is best aerodynamically, and that greater taper causes a loss of aerodynamic efficiency. Taper ratios, i.e., $\frac{\text{root chord}}{\text{tip chord}}$, of between 2 and 4 give results approximating very closely to the elliptical plan form wing, $C_{L\ max}$ being increased beyond the value for a rectangular wing of the same aspect ratio by about 5 per cent., and all drag values being decreased by amounts equal to 3 per cent. of the induced drag, plus a uniform decrease of 10 per cent. $C_{D\ min}$ throughout the range of ordinary angles.

The structural benefits of a tapered wing are readily apparent, whilst a moderate taper is conducive to good manœuvrability, though a taper ratio greater than 4 is undesirable from all aspects except structural.

During recent years the subject of tapered wings has come into prominence on account of the undesirable tendency displayed with some aircraft of "dropping a wing" when near stalling incidence. Under the heading "Induced Drag" of Chapter IV*

* See p. 49.

is shown how the tip vortex tends to reduce the effective incidence towards the wing-tip, and thus delay stalling over that portion of a wing. It will also be appreciated that if an aeroplane is caused to roll about the longitudinal axis, the effective incidence of the down-going wing is increased and that this increase is most pronounced at the wing-tip. In normal flight this results in a beneficial damping effect, but at high incidence, in the region of $C_{L\ max}$, a slight rolling motion, whether initiated by the pilot, or due to unstable air conditions, may cause a stall to take place and it will be seen that the tapered wing is more liable to develop a tip stall than one of rectangular shape.

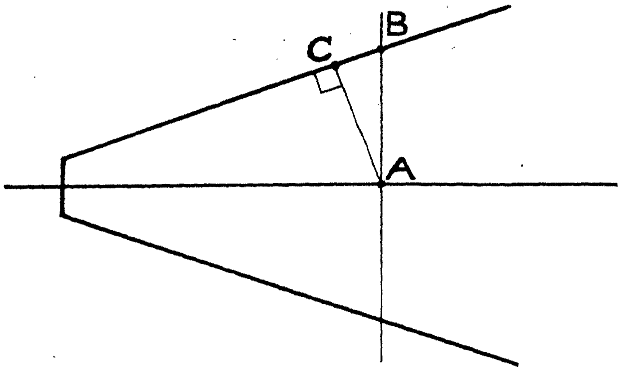


FIG. 41

Excessive taper results in a falling-off of aerodynamic efficiency, due no doubt to loss of pressure gradient. In Fig. 41 the pressure at C is sensibly equal to that at B, but being closer to A, a spanwise flow component is set up with loss of lift. This applies also to the trailing-edge, but to a lesser extent on account of the more gradual pressure gradient over the rear part of the chord. It will be noticed that exaggerated taper causes loss of lift all along the span ; in other words the " end loss " is spread over the entire span with an evening-up of the unit loading, or true C_L, from tip to tip.

The effect of this is equivalent to a shift of the vortex core inboard from the tip, with the result that the induced velocity over the outer part of the wing tends to decrease, and for excessive taper, i.e., a triangular plan shape with pointed tips, the induced velocity becomes upward at the extreme tip, thus causing stalled conditions at quite small angles of attack. The danger of early tip-stalling with such a plan form is readily seen : It may be over-

come, to a large extent at least, by giving a twist to the wing, though theoretically the geometric twist necessary to produce an elliptical C_L distribution across the span is considerable, varying from about $-13°$ for a taper ratio of 2, to $-20°$ for a ratio of 5 (the twist should not be uniform, but should increase progressively towards the tip), and this in turn causes increased induced drag, the increase for a 20° twist being roughly 10 per cent. for a ratio of 2·5, and 20 per cent. when the taper ratio is 5. On account of this it is doubtful whether a twist greater than 3° or 4° should be used, and 2° might be regarded as a preferable limiting figure.

In practice a lesser amount of wash-out than the theoretical figures given above has been found necessary due to the presence of a fuselage, or engine nacelles, which have the effect of accelerating the advent of unstable, or stalled, air-flow over the inner part of a wing.

A better method of preventing tip-stalling, or one which may be profitably employed in conjunction with a small degree of twist, is to increase the camber from root to tip, or at least over the outer sections of the wing. Alternatively, the aerofoil section may be graded along the span so that the tip section has a greater angle of maximum lift, sufficient wash-out being employed to keep the angle of zero lift constant along the span. Increase of camber results in a greater angular range of lift,* i.e., from the no-lift angle to the angle for $C_{L\,max}$, the greater angle of the latter being made use of for delaying stalling towards the wing-tips. For taper ratios up to 4, a camber grading of from, say, 2 per cent. at the root to 5 or 6 per cent. at the tip is generally sufficient for satisfactory results.

Aerofoil sections with rearward position of maximum camber,† i.e., behind the one-third chord position, give better results than forward camber locations. Rearward shift of the point of maximum camber over the tip portion of a wing is likewise beneficial in this respect.

Another solution to the tip-stalling problem, and again one that may be used in conjunction with camber variation, is provided by suitable grading of the wing thickness over the outer portion of the span, but avoiding, if possible, the rather critical region of 12 per cent.‡

* See p. 58 (Chap. V).
† See pp. 59 and 60 (Chap. V).
‡ See p. 60 (Chap. V).

Other factors introduced by taper are the decrease of R.N. and the relative increase of surface roughness towards the tip. Both are likely to cause loss of lift,* though the peak of the lift curve should become more rounded.

Wing-Tip Shape.

The shaping of wing-tips consists in reality of tapering the wing over the extreme outer portions of the span, and of course, some increase of aspect ratio takes place. The aerodynamic benefits of a well-shaped wing-tip are roughly half those obtained by the best form of taper, and, apart even from aesthetic considerations, no rectangular wing should be denied the advantages of so simple a modification.

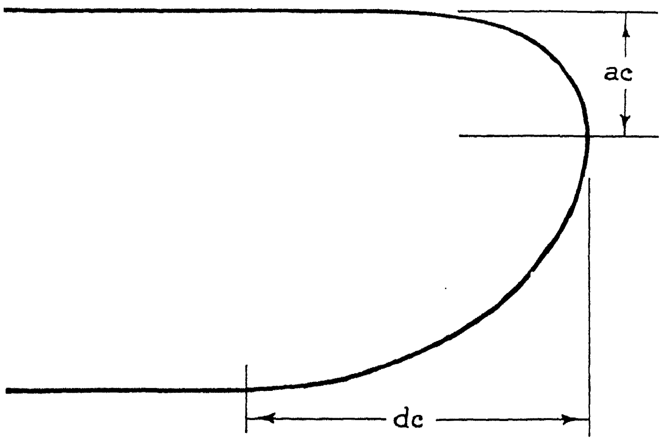


FIG. 42.—WING-TIP SHAPE

For best results the maximum span should be at a distance of $\frac{c}{3}$ from the leading-edge, the outline curving to the two edges in the form of either quarter ellipses, or circles.

Rounding of the wing-tip is accompanied by some change in the position of the C_P of the wing. If the C_P coefficient of the rectangular wing is denoted by $C_{CP\ rect}$ and using the notations shown in Fig. 42, the modified coefficient is given approximately by—

$$C_{CP} = C_{CP\ rect} - \frac{(C_{CP\ rect} - a)\,d}{10}$$

* See pp. 63 and 188.

Thus, for example, if $C_{CP\ rect} = 0{\cdot}4$, $a = 0{\cdot}3$ and $d = 0{\cdot}6$,

$$\text{Then } C_{CP} = 0{\cdot}4 - \frac{(0{\cdot}4 - 0{\cdot}3) \times 0{\cdot}6}{10} = 0{\cdot}394.$$

Effect of Sweep in Plan Form.

Wings are sometimes swept back, and more rarely forward through some small angle, for the purpose of balance, and on occasion for the sake of stability.* Such arrangements result in loss of $C_{L\ max}$ amounting to about 5 per cent. for a 10° angle of sweep, and slight increase of drag. In the case of tapered wings, backward sweep appears to accentuate the tendency towards tip-stalling, and should be used only with caution until more data on the subject become available.

A considerable amount of investigation has been carried out at the N.P.L.† on the effects on stalling and rolling stability of the inclination of leading- and trailing-edges to the lateral axis. Tip-stalling was found to occur with sweep-back of the leading-edge and only a moderate degree of taper (2·5), but considerably greater taper may be employed without early tip-stalling when the taper is limited to sweep-forward of the trailing-edge, the permissible taper increasing with decrease of aspect ratio.

Limiting values of taper for avoidance of tip-stalling and adverse rolling stability appear to be in the neighbourhood of two, or even less, where the taper is confined to sweep-back of the leading-edge, but for a swept-forward trailing-edge the taper ratio may be increased to 6 for an aspect ratio of 9, and perhaps to 8 at an aspect ratio of 6 or 7.

The explanation of the improved conditions found with the swept-forward trailing-edge is suggested as being due to the boundary layer control exerted by the inflow over the rear top surface of the wing, by which the stagnant air within the tip region is transferred to the centre of the span, with consequent straightening of the main flow at the tips, and increased stalling tendency at the centre.

The additional effect of trailing-edge split flaps‡ was observed in the experiments referred to. Where these are limited to the central 50 per cent., span conditions are little altered, but

* See p. 95.

† "Some Aerodynamic Characteristics of Tapered Wings fitted with flaps of various sorts," *R. & M.*, No. 1796.

‡ See also p. 134.

whereas the effects as regards both tip-stalling and rolling stability are adverse with a swept-back wing, the reverse is the case with a forward-swept trailing-edge. It appears that such flaps should not extend much beyond the half semi-span position and certainly should not exceed 70 per cent. of the span. Again, this applies more particularly with the swept-back leading-edge.

In this work no account has been taken of the effects of variation of thickness and camber, wing twist, and the presence of a fuselage or engine nacelles in the wing: Such factors cannot be ignored and may considerably modify the results as given above.

Effect of Twist.

The reason for, and the effect of wing twist have been dealt with at some length earlier in this chapter. The induced drag due to wash-out has been given by Lock* for an elliptic plan form, of aspect ratio 6, as $C_{Di} = 0{\cdot}000033\,\beta^2$, where β is the total wash-out in degrees. It is thus seen to increase in proportion to the square of wash-out.

During the past decade wash-out of incidence towards the tips, especially with tapered wings, has been freely resorted to for improvement of lateral stability. As has been pointed out, its place is now largely taken by adjustment of camber.

Wing Joint Gaps.

Where the span of a wing is made discontinuous for the purposes of folding, or dismantling, care should be taken to see that no gap exists in the extended wing and that there is no connecting passage between the different pressure regions above and below the wing.

Any such gap must result in adverse lift and drag changes, increasing in effect with increased area of gap. The effect is really one of aspect ratio.

Cutting away of a wing at the centre section, or next to the fuselage, for improvement of pilot's vision, produces similar results, also for the same reason, and likewise should be avoided if possible, or reduced to a minimum.

The adverse effects may be largely overcome by suitable adjustment of the camber and chord setting of such parts so as

* "Induced Drag due to Wash-out," *R. & M.*, No. 1769.

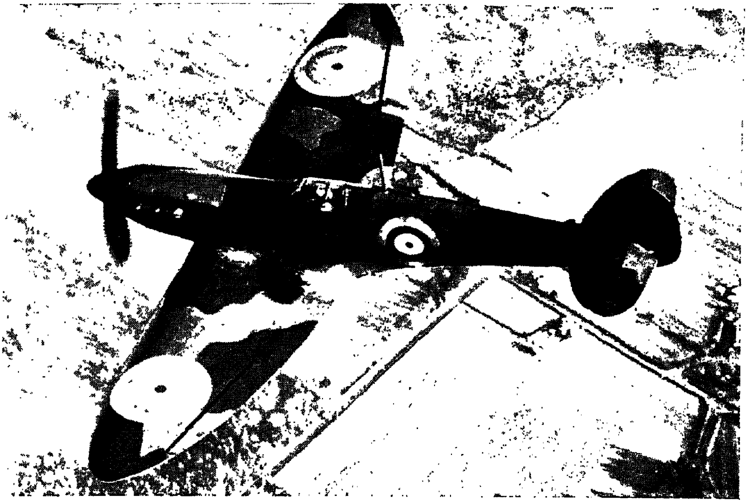


FIG. 43.—VICKERS-SUPERMARINE "SPITFIRE"
ROLLS-ROYCE "MERLIN" ENGINE
(*Reproduced by courtesy of "Flight"*)

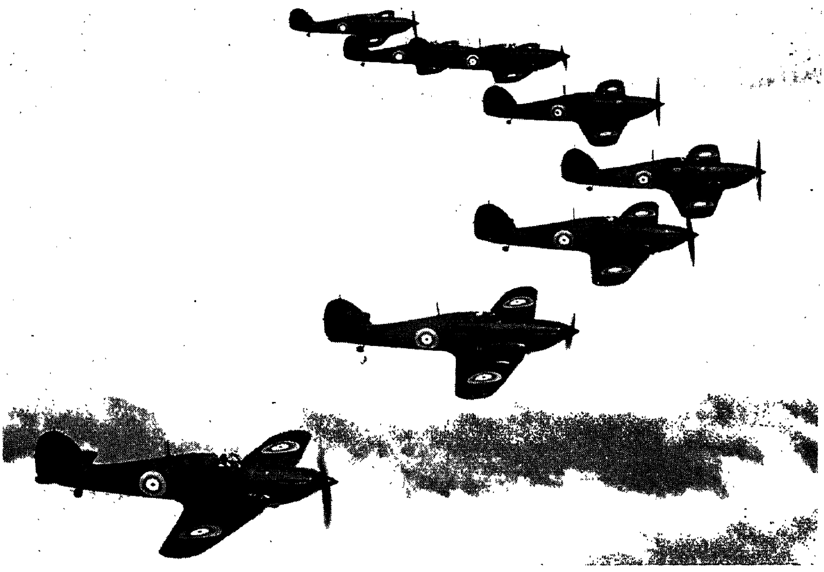


FIG. 44.—HAWKER "HURRICANES" IN FLIGHT
1050 H.P. ROLLS-ROYCE "MERLIN" ENGINE
(*Reproduced by courtesy of "Flight"*)

to avoid upsetting of the lift grading across the span. Complete reparation is not possible for all attitudes of the aircraft, and the re-arrangement of the aerofoil section and its setting should be chosen for the most important attitude of the aircraft under consideration. Interference due to protruding wing tanks should be similarly dealt with.

Wing-Tip Plates.

In order to avoid the aerodynamic losses due to wing-tips, vertical plates, or fins, have on occasion been attached to the wing-tips. Certain systematic tests have been carried out in Germany with tip shields of various shapes. As might be expected, at small angles of attack, where induced drag effects are of little importance, the end shields give additional drag with considerable increase of $C_{D\ min}$. Above a C_L value of about 0·5 there is a distinct drag decrease, depending largely on the relative area and shape of the end plate. A circular disc, of diameter equal to the wing chord, appears to give best results, with a decrease of drag of as much as one-third for equal values of lift, together with a 12 per cent. increase of $C_{L\ max}$. Increasing the length of the plate to one and a half times the chord gave slightly better drag values, but with only a 4 per cent. improvement in maximum lift.

Wing-tip discs of two different sizes, made to serve as rudders, on a British tailless aeroplane* showed increases of 5·7 and 6·7 per cent. in $C_{D\ min}$.

From the above it is seen that tip shields may be advantageous for climb, and the take-off of heavily laden aircraft, but only at the expense of curtailed top speed and general flight economy. They would be of assistance for the landing and take-off of racing machines having a small aspect ratio, but the accompanying reduction of top speed would far more than outweigh such benefits.

It would seem that for best results end plates should be inclined at some small angle to the longitudinal axis, and that the inclination below the wing should be in the reverse direction to that above.

Where end shields are required to act as rudders, as on the main wing of a tailless aeroplane, or on the extremities of the tail-plane of a multi-engined aircraft, the resulting benefits may

* *R. & M.*, No. 1577.

be greater, but the mutual interference effects should not be overlooked. Again when both rudders can be rotated outwards simultaneously, a powerful braking device becomes available.

Biplane Effects.

Biplane Interference. When two or more planes are situated one above the other, so as to form biplane, triplane or multiplane arrangements, the low pressure region over the top of the lower wing is closely adjacent to the high pressure area below the upper wing, with the obvious result that some readjustment of pressure intensity takes place, which reduces the effectiveness of both planes.

Since the top surface pressure is of greater importance than that of the under surface, the lower wing of a biplane suffers to a greater extent than the upper wing. This is of importance when considering the strength of the individual wings, whereas the total effect has to be taken account of for determining the requisite wing area, and the performance figures.

Gap. Tests made in this country with biplane wings of equal span and similar sectional shape, R.A.F.15, showed that $C_{L\,max}$ increased from 1·071, for a gap of two-thirds the chord, to 1·208 when the gap was $2\frac{1}{3}$ times the chord. $\frac{C_L}{C_D}$ (max) was similarly increased from 14·1 to 17·2, whilst $C_{D\,min}$ fell from 0·0268 for a gap/chord ratio of unity to 0·025 for a ratio of $2\frac{1}{3}$.

The figures for $C_{L\,max}$ and $\frac{C_L}{C_D}$ (max) for a gap/chord ratio of unity, which represents the average value used in practice, were 1·13 and 15·4 respectively.

Compared with monoplane figures, the effects of a biplane gap equal to the chord, with no stagger, show reductions of roughly 5 per cent. and 20 per cent. for $C_{L\,max}$ and $\frac{C_L}{C_D}$ (max) respectively, together with an average increase of $C_{D\,min}$ of about 10 per cent., and in some instances by as much as 20 per cent.

It will be realised that these depreciations of the wing characteristics are far from negligible, and must cause reduced performances for the biplane arrangement, so that it is not surprising that the present tendency favours the monoplane.

The sesquiplane arrangement, in which one wing, generally

the top, has roughly twice the area of the lower wing, is a compromise which enables the advantages of the biplane structure to be combined with somewhat improved aerodynamic characteristics, though still considerably inferior to those appertaining to monoplanes.

For best results the aspect ratio of the smaller wing should be greater than the other. Practical considerations should be taken as the only limit to this rule.

The equivalent "swept" area* of a biplane may be taken as roughly $S' = gb + \frac{\pi}{4} b^2$, where g is the gap, and the coefficient of induced drag is $C_{Di} = \dfrac{2\, C_L{}^2\, S'}{b^2\left(\pi + 4\dfrac{g}{b}\right)}$.

The distributions of loading over the two planes of a biplane are dealt with in Volume II, Chapter VI.

STAGGER.—Stagger affects chiefly $C_{L\ max}$, which is increased about 5 per cent. for a 30° stagger, and in proportion for smaller amounts, with corresponding decreases for negative stagger. The effects on $\frac{C_L}{C_D}$ and $C_{D\ min}$ are negligible.

It should be realised that stagger has the effect of increasing the length, weight and drag of all interplane connecting struts and wires, especially the former, so that it is doubtful whether there is any net gain of efficiency resulting from stagger, and it is therefore employed chiefly for the improvement of the pilot's view, or for adjustment of the C_P location.

The effect of stagger on the loading of the two wings of a biplane is of some consequence. Negative stagger, that is when the top wing is positioned behind the lower wing, results in greater losses of the upper wing lift and enhanced efficiency of the lower wing. This follows naturally from a consideration of the disturbed air-flow due to interference.

There is little difference in the proportion of lift load carried by the two planes when set at any degree of negative stagger. Fig. 45 shows the distribution of lift on the wings of a biplane with positive, zero and negative stagger.

DECALAGE.—Decalage, as applied to the wings of a biplane, is defined as the difference in the angles of incidence of the two planes, being reckoned as positive when the setting of the

* See p. 51.

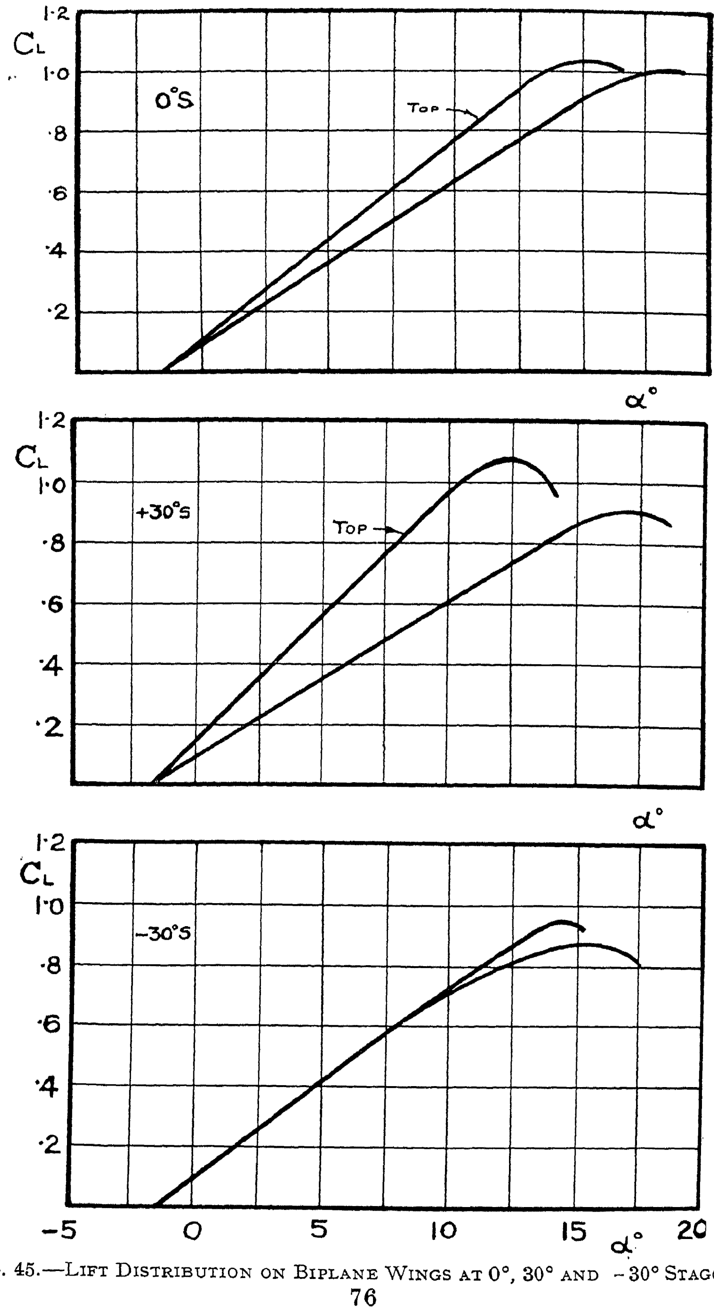


FIG. 45.—LIFT DISTRIBUTION ON BIPLANE WINGS AT 0°, 30° AND -30° STAGGER

top plane is greater than the setting of the lower. Information on the effect of employing slight positive or negative decalage seems somewhat conflicting, and it is doubtful whether it is attended with material gain in C_L or $\frac{C_L}{C_D}$ values, although obviously, the relative loading of the two wings must be to some extent dependent on the decalage setting.

Some stabilisation of the total C_P position is possible by the employment of decalage.

In a few instances the upper and lower wings of a biplane have been of different section. No appreciable aerodynamic gain or loss appears to accompany such arrangements.

The term decalage is also used to denote the relative setting of the tail-plane, and main plane, or planes.

TRIPLANE.—The triplane is very seldom used on account of the inferior aerodynamic characteristics inherent in such an arrangement. As would be anticipated, the losses induced with a triplane are approximately twice those of a biplane, being about 10 per cent. for $C_{L\,max}$ and 30 per cent. for $\frac{C_L}{C_D}$ (max), as compared with monoplane figures.

The load proportions on top, middle, and bottom wings are roughly 37, 30, and 33 per cent. respectively.

TANDEM PLANES.—All aeroplanes, except tailless, have tandem wings. The rear plane, or tail, is generally small in comparison and is set to take little or no load in normal flight. The " tail-first " aeroplane employs a lifting forward stabiliser, which is so set that it stalls before the main wing, and thus makes stalling of the latter almost impossible, though the interference effect, principally induced drag, seriously affects the general performance.

The true tandem arrangement is one in which there are two main lifting surfaces, set one behind the other, and several attempts have been made to exploit the possible advantages that such an arrangement may offer.

The decalage angle (reckoned positive when the front wing incidence exceeds that of the rear member) is of considerable importance in such an arrangement. The *effective* decalage angle is increased from the apparent decalage by the downwash angle due to the passage of air-flow over the front wing, which,

of course, does not remain constant throughout the range of angles of attack.

Most tests made with tandem aerofoils have incorporated positive apparent or effective decalage, which means of course that the forward wing supplies more than half the total lift. Such an arrangement gives a stable C_P movement, due to the greater relative increase of lift of the rear aerofoil, and good longitudinal stability qualities result.* If, however, the optimum value of $\frac{L}{D}$ is sought the wings must be set at negative decalage, and for the greatest $C_{L\ max}$ for the combination, means of adjusting the geometric decalage during flight should be provided. The longitudinal stability is now lost.

Another favourable quality of the tandem wing aircraft is the reluctance to stall of the rear wing due of course to the front wing downwash, this being a feature of the "tail-first" machine as already explained. Recent experiments have shown the apparent stalling angle of the rear wing to be in the neighbourhood of 22 or 24 degrees.

The one factor that almost condemns tandem aerofoils is induced drag. The rear wing, acting as it does in the downwash of the forward wing, has its true lift axis set at an increased rearward inclination, with a consequent greater induced drag component. Rearward shift of the rear wing decreases the adverse effects, but is accompanied by structural complications. If the geometric decalage is fixed, and is positive, the aerodynamic efficiency is poor, whilst if the true decalage is adjusted for maximum efficiency, stability is sacrificed. The addition of a third, or tail, plane for stability results in a still heavier induced drag penalty.

The performance of a tandem combination, with a gap between the wings of 2 *c*, is roughly equal to that of a biplane, but decreases slightly as the gap is lessened.

The Autogyro.

In this type of aeroplane the fixed lifting surfaces are replaced by revolving blades which form a windmill. The blades, 2, 3 or 4, are mounted to rotate about an axis that is roughly vertical, but which is tilted backward from the normal to the relative air-flow.

* See p. 91.

Forward speed is obtained by means of a normal airscrew, the relative air-flow thus generated causing the windmill blades to rotate, with the production of lift and drag, but torque as applied to airscrews* must be zero of course.

It will be noticed that the absolute speed of a blade will vary during each revolution, being greatest at the mid-point of the forward arc, and decreasing over the retreating path. If the blades were rigidly attached to the hub this would result in uneven lift across the blade disc, and tilting of the aircraft. To avoid this the blades are allowed to hinge on horizontal pivots and thus ride up and down, or flap, during each revolution, with consequent evening-up of the lift.

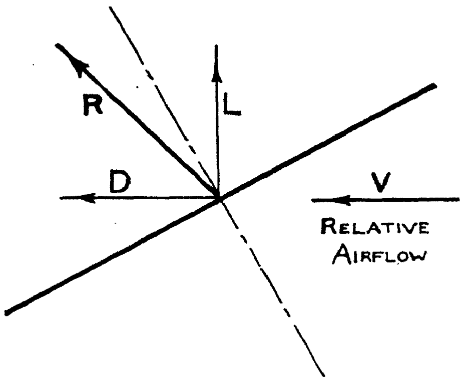


FIG. 46.—FORCES ON AUTOGYRO DISC

Control about any or all axes is obtainable at the pilot's will by tilting of the main axis.

The blades possess a large degree of flexibility along their span, the centrifugal loading being relied on very largely to resist bending due to the aerodynamic loading. The area of the blades divided by the disc area is referred to as the solidity ratio, the present-day average value being about 0·07.

The aerodynamic reaction on the blades is inclined rearward from the normal (see Fig. 46) and gives components of lift and drag, L and D. The coefficients of these forces are based on the disc area, $S = \pi r^2$, and the forward velocity of the aircraft, a maximum value of about 1·0 for C_L being obtainable at a disc incidence in the neighbourhood of 40°. But at this

* See p. 155.

incidence the $\frac{L}{D}$ ratio is little greater than unity, and obviously thrust must be augmented by loss of altitude. Descent is possible along a vertical path, but the speed is greater than the vertical component in a glide, at, or near, the attitude for $C_{L\ max}$, owing to the blades being stalled.

The ratio of $\frac{L}{D}$ for the rotor increases with forward speed, and is a maximum—about 8—just below top speed. Decrease of rotor drag for increased speed, is accompanied by greater parasite drag which thus prevents the attainment of a nett $\frac{L}{D}$ much higher than 5. At the best climbing attitude the rotor drag is relatively high, and again the $\frac{L}{D}$ of the whole aircraft is about 5, or considerably inferior to the orthodox aeroplane.

Really high-speed autogyro flight does not appear possible, since a smaller disc diameter means a greater disc incidence, with loss of $\frac{L}{D}$, whilst a decrease of the solidity ratio must be accompanied by greater rotor speeds, in which case the centrifugal loading tends to become excessive.

In the jump-start autogyro the blade incidence is variable. Thus the blades are set in the position for $C_{D\ min}$, and are speeded up by the engine. By increasing the blade angle of attack, considerable lift is obtained from the stored energy of momentum, and the aircraft leaps into the air. This is followed by a gentle glide to obtain forward speed, when the climb may be continued.

The Helicopter.

The helicopter is a craft in which the lift is obtained by means of engine-driven blades, or screws, revolving in a horizontal plane. Forward motion is obtained by tilting the rotor axes forward. In order to avoid tilting due to uneven lift across the rotor disc, and torque effect on the body, two screws are generally employed with opposite direction of rotation.

The Sailplane.

The sailplane,* or high-efficiency glider, is an aeroplane

* See *Sailplanes* by the same author.

devoid of mechanical power, and which depends on the kinetic energy present in the air for sustentation in flight. In other words, the sailplane is unable, without external assistance, to remain in flight without loss of height in perfectly calm air. The most elementary, and most easily understood, source of lift, made use of by the soaring bird and sailplane alike, is the up-current caused by a wind blowing against a hill, or other obstacle. If the vertical component of the velocity of the air is equal to, or greater than, the sinking speed of the sailplane, height can be maintained, or gained. This is known as "static" or hill soaring.

Thermal movements of the air, due to the effects of solar rays, provide vigorous vertical convection currents, and so provide a source of energy available under almost all geographical or physical conditions. Still another type of soaring flight, termed "dynamic," is possible when the horizontal movement of air varies vertically (ground friction), or horizontally (gusts), for by climbing into a wind of increasing velocity the loss of speed due to gain of height is made good by the higher speed of the air. In practice such climbs are of short duration, and are generally followed by down-wind dives, converting energy of height into energy of speed, prior to the next climb into wind.

The one consideration, or at least, the main requirement, in sailplane design is aerodynamic efficiency, and it may truly be said that present-day sailplanes provide the nearest approach to the "ideal" aeroplane.* Except for a fuselage of almost perfect streamline shape and small tail organs, the sailplane consists of a high-efficiency wing, designed for low induced drag. Due to the fact that sailplanes make use of coarser angles of incidence than is usual with most other aeroplanes, the factor of induced drag becomes paramount. Consequently, a high aspect ratio is imperative, a figure of 20 being quite common, and 26 not being unknown.

The high aspect ratio thus dictated results in a relatively heavy structure, but this in turn is compensated for by increased main plane area. Carrying this to the extreme, it will be seen that the parasite drag coefficient becomes almost insignificant, and a value of $\frac{L}{D}$ approaching that of the maximum for the

* See p. 210.

wing alone becomes possible, with a correspondingly fine gliding angle. $\left(\text{Tan } \theta = \frac{D}{L}\right)$

Such factors, however, as cost of manufacture, housing, transport and launching, and more particularly of manœuvrability, especially for circling in the limited areas of thermal lift, restrict the span to about 50 ft. or perhaps 60 ft. in special cases, for single-seat sailplanes.

The low rate of sinking speed, obviously a desirable feature of the sailplane, can be achieved by a large expanse of wing with consequent light wing-loading, but this also infers a low forward speed. A low speed is disadvantageous for long distance flights, for which higher speed and a fine gliding angle are more favourable qualities. Moreover, a relatively high speed enables rapid flight from one source of lift to another to be made. The tendency, therefore, is towards machines of somewhat curtailed span, for manœuvrability, but of high aspect ratio for low induced drag, whilst the resultant diminished wing area provides for a high speed of flight.

The Tail-First Aeroplane.

An advantage claimed for the tail-first aeroplane is its immunity from stalling. The stabiliser may be so set that it reaches its $C_{L\ max}$ attitude before the main plane, so that further increase in the angle of attack causes its lift to decrease, with the accompanying lowering of the forward plane.

It will be noticed that the apparent longitudinal dihedral angle may be zero, or even negative, but the tail-plane downwash has to be allowed for. This means of course that the maximum main-plane C_L cannot be made use of in flight, and results in higher landing speeds.

The downwash from the forward plane causes disturbed flow over the main plane and of course results in a decreased effective angle of incidence over the central portion. Since this change of incidence is not constant throughout the flight range, it cannot be easily allowed for by, say, variation of aerofoil section, or wash-out at the wing-tips, and moreover it is likely to lead to early stalling of the outer portions of the main wing. Loss of aerodynamic efficiency is difficult to avoid in such an arrangement.

CHAPTER VII

THE FORCES ACTING ON AN AEROPLANE IN STEADY FLIGHT

The four main forces present during flight are :—

(*a*) Total lift, L,
(*b*) Weight of aircraft, W,
(*c*) Airscrew thrust, T, and
(*d*) Total drag, D.

Of these (*b*) and (*d*) are always present. The only force that remains sensibly constant throughout is (*b*), although this is subject to certain variation due to consumption of fuel, and possibly the release of loads. Lift is present during all phases of flight, with the possible exception of a vertical dive, and momentarily during changes of attitude in manœuvres. Thrust is provided by the engine, through the propeller, and is varied at the will of the pilot, and to some extent, by the flight attitude.

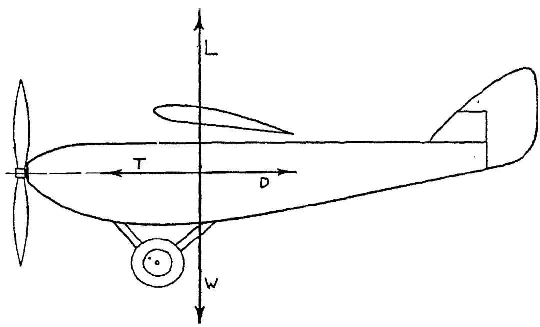


FIG. 47.—THE MAIN FORCES ACTING DURING STEADY FLIGHT

NORMAL HORIZONTAL FLIGHT

There are two possible arrangements of the four forces in normal horizontal flight. Either all four forces are coincident, as in Fig. 47, or there are two couples formed by L with W,

and T with D, the couples being of equal intensity, but of opposite sense to provide equilibrium (Fig. 48). In both cases, L must be equal to W, and T equal to D.

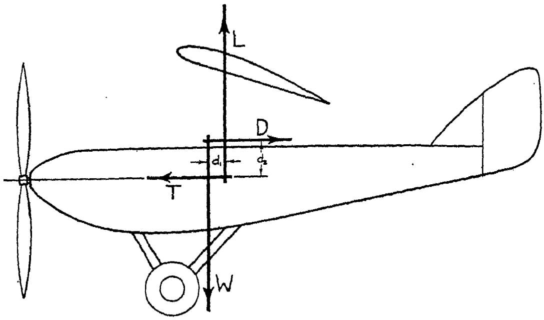


FIG. 48.—THE MAIN FORCES ACTING DURING STEADY FLIGHT

It is desirable that when thrust is no longer operative, whether by intentional use of the engine controls, or through engine failure, there should be a nose-diving tendency, i.e., the resultant moment due to the $\frac{L}{W}$ couple, and the moment of D about C_G should tend to depress the nose of the aircraft. Thus, in Fig. 49, if D is assumed to act at a distance d_3 above C_G then $Ld_1 - Dd_3$ should be positive.

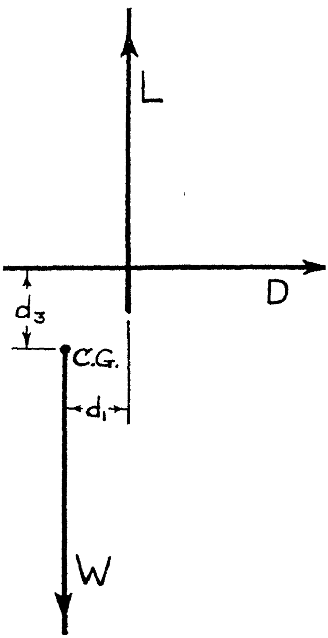


FIG. 49

It should be noted that when thrust ceases to act both L and D tend to decrease, and this may be accompanied by some movement of the C_P, the effects of which should be allowed for.

The ideal arrangement appears to be obtained when the line of thrust is below the C_G so as to ensure an increased nose-dive moment when the engine ceases to supply thrust. This is probably most easily achieved by the high wing monoplane

arrangement, whilst the introduction of retractable landing gear makes the low thrust-line easier of attainment.

The sudden loss of one force of a system is almost bound to upset equilibrium momentarily, a condition that cannot always be avoided, but by arranging for a small down tail load in normal flight, which will largely disappear with slipstream (at least to a greater proportional extent than the main plane lift) a nose-down tendency may be secured.

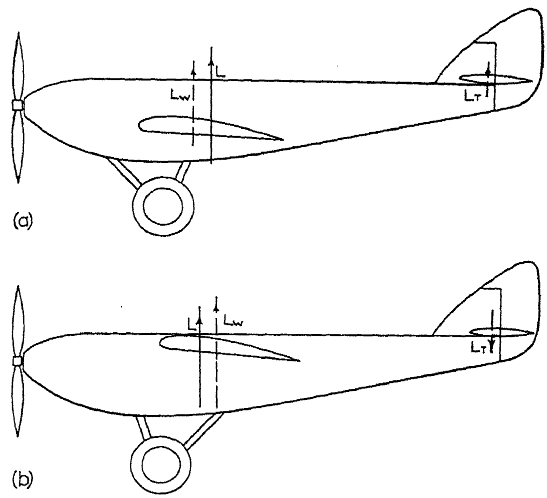


FIG. 50.—LIFT COMPONENTS ON WING AND TAIL

The total lift, L, is made up of the wing lift, L_W, and tail lift, L_T. If the main wing C_P acts in front of L, then the tail load acts upwards, or helps to support W, i.e., $L = L_W + L_T$ (Fig. 50 (*a*)). But if the wing C_P is behind the position of L, then there is a down-load on the tail, whilst the wings have to support a load in excess of the total weight, or $L_W = W + L_T$ (Fig. 50 (*b*)).

A down-load of any considerable magnitude on the tail, although often necessary for stability, is undesirable, since it constitutes a parasite load, decreases the disposable, or pay-load, and tends to affect adversely both the maximum and minimum speed. An excessive tail load tends moreover to restrict the manœuvring powers of an aircraft and should therefore be avoided.

2. GLIDING AND DIVING—ENGINE OFF

Here three forces are present, W, L and D ; L and D together give R, the vertical aerodynamic resultant, which must equal W.

The force producing movement along the inclined flight path is a component of the weight, and is given by W sin θ, ere θ is the angle made by the flight path with the horizontal. Hence W sin θ = D, and W cos θ = L.

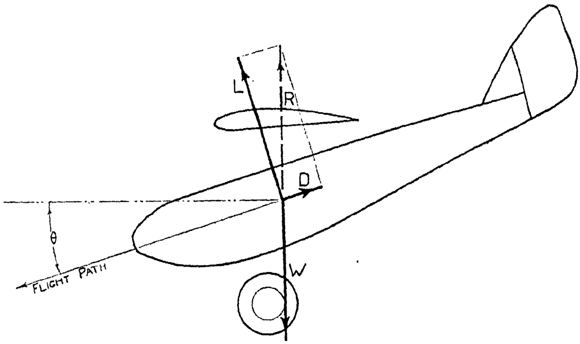


FIG. 51.—FORCES ACTING IN A GLIDE

As the glide becomes steeper, and the speed exceeds that of normal horizontal flight, it is referred to as a dive. The weight is balanced by components of lift and drag, or W = L cos θ + D sin θ. Both D and θ increase with the steepness of the dive, and therefore L decreases. Due, however, to the rapid backward shift of the C_P on most aerofoils at the finer angles of incidence, a down-load becomes necessary on the tail for equilibrium, and the actual wing lift may still be relatively heavy (see Fig. 52), and when it is remembered that under this condition the wing is subjected to

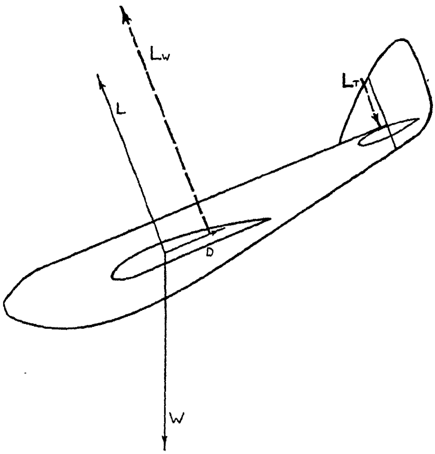


FIG. 52.—LIFT COMPONENTS IN A DIVE

a down-loading over the forward part and an upward load over the rear, it is realised that this up-load component may be very considerable.

3. CLIMBING

CASE I.—If the line of thrust coincides with the path of flight, Fig. 53, then $T = D + W \sin \theta$
and $L = W \cos \theta$.

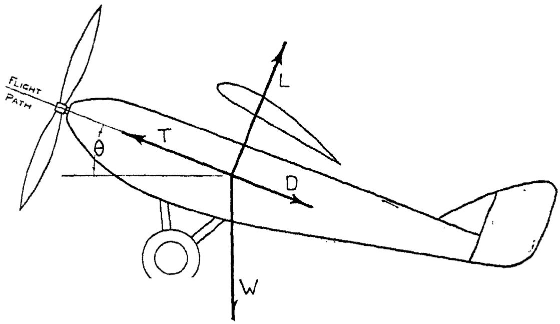


FIG. 53.—MAIN FORCES ACTING IN A CLIMB

CASE II.—If the thrust line is at an angle ϕ to the flight path, Fig. 54,

$$\text{Then } T \cos (\theta - \phi) = D + W \sin \phi$$
$$\text{and } L = W \cos \phi - T \sin (\theta - \phi).$$

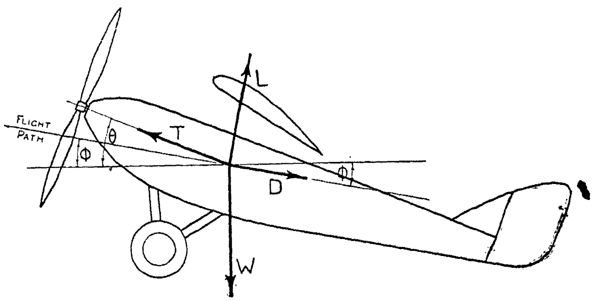


FIG. 54.—FLIGHT FORCES ACTING IN A CLIMB

Forces on Flying-Boat in Horizontal Flight.

The case of the flying-boat constitutes an extreme case, due to the fact that the airscrews and engines must be placed

well above the water, with a consequent large thrust/drag couple that is difficult to avoid. Balancing by the lift/weight couple means placing the lift well forward of the C_G, which in turn infers a down tail load under horizontal flight conditions. When the engines are switched off, or throttled, during flight there is a diminution of the T/D overturning couple, and an increase in the nose-up L/W couple due to forward movement of the wing C_P, with consequent danger of stalling.

The desirable rearward movement of L may be achieved by decreasing the tail down-load, or even by its reversal to an up-load, and this may be arranged for in flying-boats by inclining, or up-setting, the engines so that the slipstream increases the effective negative incidence of the tail-plane while the engines are running, the effect of which disappears immediately the airscrew thrust ceases to be operative. The alteration of the air speed over the tail-plane with engines-on, and engines-off, also contributes towards the necessary change in the tail force required for balance.

The down-load on a tail-plane in flight is an unsatisfactory feature since the lift of the main planes has to be increased by a corresponding amount. Roughly speaking, every 150 lb. of down tail load means the carrying of one passenger less with no corresponding saving of engine power and fuel.

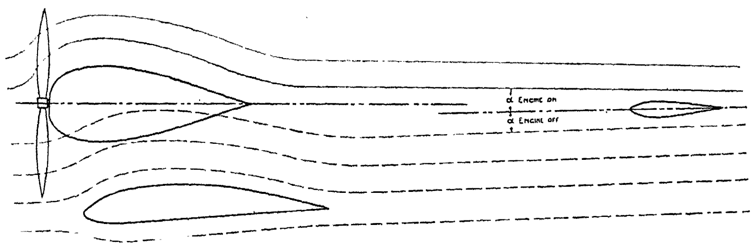


Fig. 55.—Longitudinal Stability of Flying Boat by Up-setting of Engine

4. TURNING

For the execution of a turn it is necessary that a force acting towards the centre, or pivot, of the turn should be introduced. This is obtained by banking the machine so that the lift reaction on the wings is inclined inward, and thus provides a horizontal component, as shown in Fig. 55. This is called the centripetal component of the lifting force. The air coming within the in-

fluence of the aeroplane is now deflected outward as well as downward, so that the outward or centrifugal force exerted on the air is equal to the centripetal reaction on the main planes which diverts the machine from its straight course.

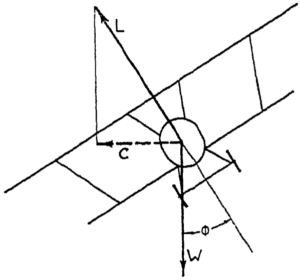


FIG. 56.—FORCES ACTING IN A TURN

For the aeroplane to continue in the same horizontal plane, the vertical component of lift must be equal to the weight, or $L \cos \phi = W$.

The centripetal component ($C = L \sin \phi$), is an unbalanced force causing motion in a circle, that is, a continuous acceleration towards the centre of the turn.

Centripetal force $= \dfrac{W V^2}{g r}$, where r = radius of turn,

Hence $\dfrac{W V^2}{g r} = L \sin \phi$

but $W = L \cos \phi$

$$\text{Therefore } \tan \phi = \frac{V^2}{g r} \tag{18}$$

It is noticed that, for horizontal flight, the value of L must increase during a turn and its magnitude is given by $L = \dfrac{W}{\cos \phi}$

Equation (18) shows that there is a correct angle of bank, ϕ, for any given radius of turn at any one velocity, and furthermore that the angle is not influenced by the weight of the aircraft.

CHAPTER VIII

FLIGHT STABILITY

The Three Main Axes of Movement.

Flight movements take place in three dimensions and there are, therefore, three mutually perpendicular axes about which an aeroplane may be considered to rotate. These main axes pass through, and are concurrent at, the centre of gravity. Aircraft, generally, are supplied with three distinct control systems which govern movement about these axes.

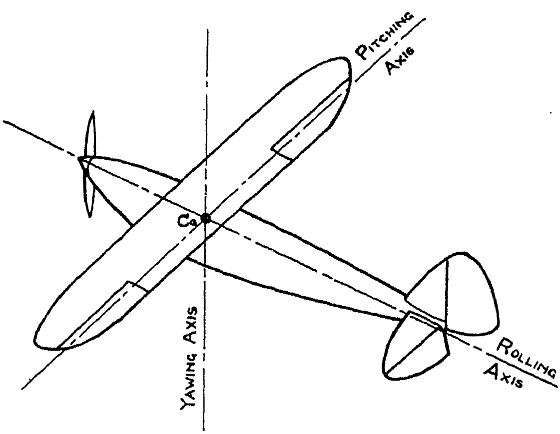


FIG. 57.—MAIN AXES OF MOVEMENT

The *longitudinal,* or *rolling,* axis (see Fig. 57) is the horizontal centre line passing from the front of the aeroplane to the rear, through the centre of gravity. Rotation about this axis causes one wing-tip to rise whilst the other falls, such movement being controlled by the ailerons.

The *lateral,* or *pitching,* axis is parallel to a line joining both wing-tips; movement about this axis being brought about by use of the elevators.

The *normal,* or *yawing,* axis is the vertical axis about which

an aeroplane rotates for a change of direction, the other two axes being in the horizontal plane : Such change is effected by means of the rudder.

Every aeroplane should be capable of being controlled by the pilot about the main axes under all conditions of flight. Without this provision, an aircraft would be considered unsafe to fly. However, in order to relieve the pilot of continuous controlling operations during flight, most machines are designed to give inherent stability about the three main axes for conditions of normal straight flying.

In certain flight conditions aeroplanes are generally unstable, and are kept within control solely by means of the control systems.

Meaning of Stability.

If an aeroplane, or any other vehicle of movement, is moving under a steady state of conditions, and is diverted momentarily from its path by some external force, such as a local air disturbance, it will tend to return to its original state and direction relative to the medium of support, only if it is stable. This is generally referred to as " inherent " stability.

On the other hand, if the initial deviation causes a continuation of the movement, then the aircraft is said to be unstable about the axis, or axes, of movement, under the conditions obtaining.

There is a third possible state of affairs, known as neutral stability, in which there is no tendency for a movement, or change of direction, once started, to continue, unless the force producing that change remains operative. But neither is there any restoring force brought into play to return the machine to its original state.

The whole question of stability of aircraft is extremely involved, owing largely to the interconnection between movements about the three main axes, and provides, perhaps, one of the most difficult of all aircraft problems. Only a simple, general explanation will be attempted here.

1. Longitudinal Stability.

By longitudinal stability is meant stability about the lateral axis, the movement being in the plane of the other two axes: In other words, it is the tendency for an aeroplane to remain

on a level (or inclined) path without prolonged up or down deviation.

For instance, consider an aeroplane in straight horizontal flight, and assume for the present that the C_P is immediately over the C_G so that L and W balance with no force on the tail. This means that the tail-plane axis is parallel to the lines of air-flow (assuming a symmetrical tail-plane section). Note that the tail-plane appears to be set at a positive incidence, and, in fact, is, relative to the datum line of the aeroplane, but the downwash from the main planes cannot be ignored.

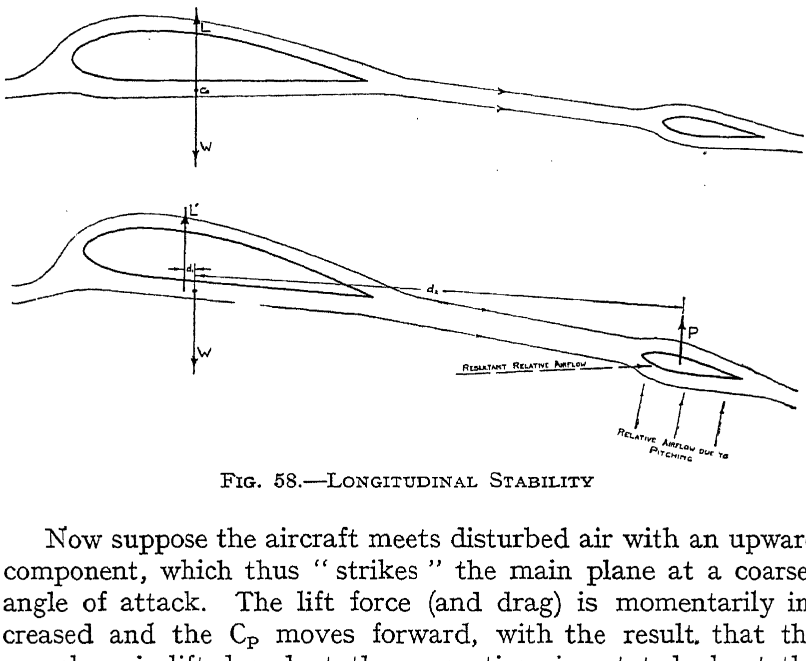


FIG. 58.—LONGITUDINAL STABILITY

Now suppose the aircraft meets disturbed air with an upward component, which thus " strikes " the main plane at a coarser angle of attack. The lift force (and drag) is momentarily increased and the C_P moves forward, with the result that the aeroplane is lifted and at the same time is rotated about the C_G so as to depress the tail.

Whilst the downward rotation of the tail continues the relative up-wind (see Fig. 58) has to be combined with the original air-flow to obtain the resultant flow over the tail, and it is now seen that the tail makes a positive angle of attack with this resultant flow. This angle is further increased due to the air-flow from the main plane not taking so steep a path, *relative to the aeroplane's axis*, as previously.

If P d_2 is larger than L′ d (Fig. 58) the original rotation will

be counteracted and reversed. Further, the increased drag causes the forward speed, and thereby the lift, to diminish, which roughly compensates for the falling-off of P as the nose goes down. If over-correction takes place, the motion is damped down by a reversal action of the tail-plane.

It will be seen that if the tail-plane had been set so as to give an initial up- or down-load in normal flight, the above explanation still holds good.

Angle of Downwash at Tail-Plane.

The angle of downwash at the tail-plane, due to the main plane, is approximately $\epsilon^\circ = 35$ (19)

2. Lateral Stability.

If an aircraft becomes tilted over to one side, i.e., if it rotates about the longitudinal axis so that one wing drops, there should be a tendency for it to return to an even keel; otherwise it is not laterally stable.

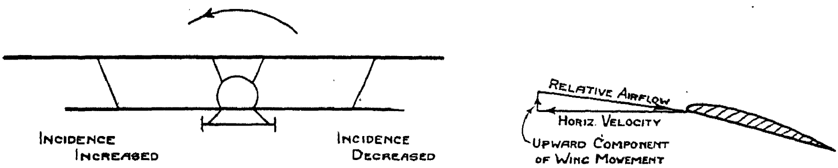


Fig. 59.—Effect of Roll

It will be realised that a rotation in the manner outlined causes a temporary increase in the effective angle of attack of the down-going wing, and so results in an increased lift, whilst on the other wing there is a corresponding decrease of lift. Thus a righting couple is set up which tends to arrest the upsetting movement. This righting effect disappears immediately the rotation ceases and is incapable of returning the machine to its original position. In fact it tends also to prevent restoration, and does not therefore give true stability.

Methods of obtaining Lateral Stability.

Fig. 60 shows an aeroplane tilted at some angle to the horizontal. It is noticed that the direction of the lift force is also inclined and is no longer in the same line as the action of W, so that a side-slip component is introduced. This is accompanied by a

side pressure on the fuselage, wing, and other parts, and, if the centre of this side pressure is above the C_G there will be a righting couple brought into play. Note that the keel surface here is not merely the "silhouette" surface, but includes *all* side surfaces exposed in the direction of the side-slip, e.g., twin rudders give greater keel surface than a single rudder, whilst the spanwise

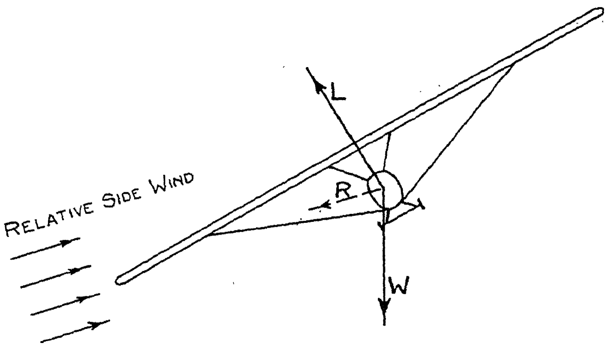


FIG. 60.—LATERAL STABILITY

inclination of the wing to the direction of side-slip increases its projected area. It will also be noticed that the relative side wind slightly reduces the angle of incidence of the wing, which causes some loss of lift and brings the direction of side-slip more nearly into line with the lateral axis.

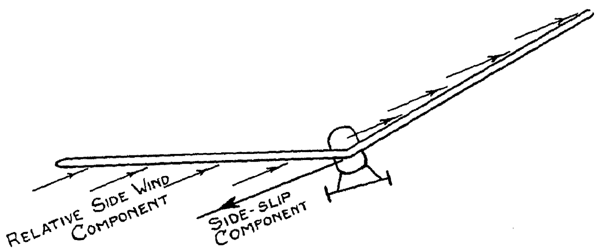


FIG. 61.—EFFECT OF LATERAL DIHEDRAL

The desired effect may be obtained by any one, or combination of the following:

(*a*) By placing the wings high relative to the body, and thus raising the C_P of the keel surface (see Fig. 60).

(*b*) By the inclusion of a high fin, or rudder, or the addition of an extra fin above the wings, or fuselage.

(*c*) By setting the wings at a lateral dihedral angle, which

raises their average height to some extent, but also, and of chief importance, results in a larger effective angle of attack on the down-going wing than on the raised wing (see Fig. 61), thus producing the necessary righting force.

(*d*) By giving sweep-back to the main planes in plan shape, so that the effective span is increased on the lower wing with a consequent gain in lift, and decreased on the higher wing, which again produces a righting couple.

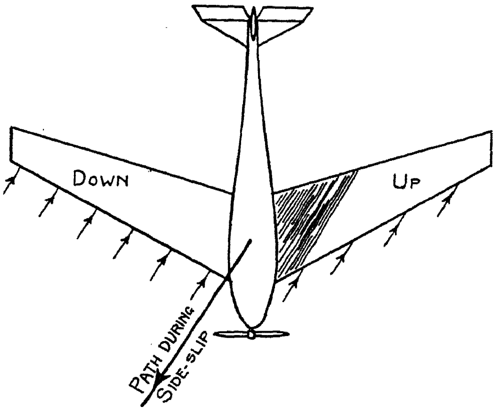


FIG. 62.—EFFECT OF SWEEP-BACK

Note also that the air-flow over the wing roots is upset by the presence of the body, but more pronouncedly on the upturned wing. Sweep-back has the effect of increasing dihedral whilst sweep-forward has the opposite effect.

The method generally adopted for providing lateral stability is the employment of lateral dihedral; most biplanes and low-wing monoplanes being thus designed, whilst the high-wing effect, (*a*) above, is generally relied upon for high-wing and parasol monoplanes. The high-wing is sometimes said to give a "pendulum" effect, but this applies no more here than in the other arrangements, since without side-slip and side pressure there will be no righting effect.

Extra vertical fins, (*b*) above, have sometimes been employed with flying-boats, where the large keel surface of the hull provides a low setting of the side area C_P.

Low-wing monoplanes have been found to give a good degree

of lateral stability without incorporating lateral dihedral, for although the side surface C_P is relatively low, so also is the C_G of such an arrangement (Fig. 63).

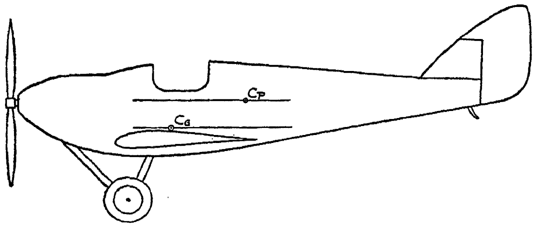


FIG. 63.—SIDE SURFACE C_P

Sweep-back is seldom, if ever, made use of primarily for the sake of lateral stability, for it has been found that 1 degree of dihedral gives roughly the same stability effect as 10 degrees of sweep-back.

Where, as is usual, the side surface C_P is behind the C_G, there is an additional effect by which the nose of the aeroplane tends to turn towards the down-wing side, and this should be counteracted by means of dihedral. In all cases the lateral stabilising effect of the wing should be roughly equal to that of the vertical tail, or preferably slightly greater owing to the relatively large leverage of the tail about the C_G.

It will be noticed that in the neighbourhood of stalling incidence, main plane dihedral may have the reverse to the desired effect (autorotation), whilst the vertical tail C_P is lowered in this condition and may be below the C_G, so that in the absence of some auxiliary lateral stabilising device the degree of such stability is likely to be small and even negative. The beneficial effect of a high fin and rudder is apparent in this respect.

3. DIRECTIONAL STABILITY.

As the name implies, a machine is directionally stable if it tends to keep on a straight course, without turning off to either side when meeting with local air disturbances.

Suppose some air eddy, or other disturbance, has caused the aeroplane to swing a few degrees off its course. Owing to its momentum, the machine tends to travel along its original path and hence a side-slipping movement is introduced, which would be

detected by the pilot in an open cockpit by the increase of pressure on one side of his face. Every exposed part of the aircraft receives the air-flow (relative) at some angle to its fore and aft axis, which thus sets up a sideways reaction, made up of the separate reactions of all such parts (Fig. 64).

The chief components on which this effect is felt are the fuselage; fin and rudder; interplane, or lift, struts, and landing chassis. Every area in front of the C_G exerts a turning moment tending to increase the angle of yaw, whilst all surfaces behind the C_G give moments of opposite sign. The reaction of parts, such as the fuselage and engine nacelles, having continuous surface both in front of, and behind, the C_G, acts strictly at the C_P of the side surface, which may generally be taken at one quarter of the distance from the leading-edge of each component.

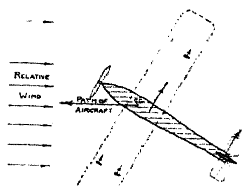


Fig. 64
Directional Stability

Directional stability is obtained by ensuring that the moment of areas behind the C_G, but neglecting that of the rudder, is greater than the total moment in front, and this is generally achieved by the employment of a vertical fin, of suitable size, above the rear fuselage. Obviously the further back this fin can be placed, the smaller it need be, on account of the greater lever arm. Structurally the usual fin position provides for a good rudder attachment and in some cases the fin is used also as a king-post for bracing the tail-plane.

It may be noted also that where the pressure on a side surface is spoken of, this really means the difference in pressure on the two faces, since the behaviour of the air-flow over a fin, or strut, is identical with that over an aerofoil (in fact they are vertically-placed aerofoils) and the reaction is really a sideways "lift."

In certain full-scale experiments,* with wool tufts attached to the leading-edge of the rudder, it was noted that "in some conditions of the spin, tufts near the leading-edge of the rudder *point forward and wrap round the leading-edge*, showing that the stagnation point, where the flow bifurcates, has moved further down the chord than the point of attachment of the tufts."

* *R. & M.*, No. 1494, "Airflow about Aeroplanes shown by Wool-tufts." Melvill Jones and Haslam.

This is really only what should be expected if the rudder is considered as an aerofoil.

Sweep-back of the main planes, if of sufficient amount, combined with wash-out, or decrease of incidence towards the wing-tips, provides longitudinal stability at coarse angles. For if the wing, already at a large angle of attack, meets air, say, with an upward velocity component, the inner parts of the planes pass their $C_{L\ max}$ attitude, and the lift over these parts decreases, whilst increased lift takes place over the outer sections. Thus there is a rearward move of the C_P and a diving moment is induced.

An aeroplane incorporating both sweep-back and wash-out is almost immune from involuntary stalling, and furthermore, since the ailerons are generally situated near the wing-tips, they remain effective when the inner parts of the wings are stalled, so that involuntary spinning also is prevented, whilst even intentional spinning has been found to be almost impossible with some such designs.

Inter-Axial Effects of Stability.

It was noticed that the lateral stabilising effect is brought about by the introduction of a slight side-slip, and that therefore there must be a tendency for the nose to drop (directional stability). This could be overcome by the pilot with use of rudder, but this means that stability is not completely automatic.

If the pilot allows the turn to develop before exercising his control, the outer wing will have risen still further, due to the greater speed of that wing over its longer path, while the inner wing will have fallen still more, and the pilot must now correct with use of both rudder and ailerons. The use of controls is accompanied by increased drag, and consequent loss of speed, and hence longitudinal stability and control are also brought into account.

During the recovery of direction, the outer wing moves faster than the inner wing, which produces a rolling motion about the longitudinal axis, and so lateral stability becomes involved, which, as has already been seen, introduces other complications. Or again, if the C_P of the side surfaces is above, or below, the C_G a directional righting movement is accompanied by a lateral turning movement, and thus lateral stability is brought in here also (see Fig. 63).

Thus it is seen that the axial stabilities, which have been dealt with as simple movements about separate axes, are really inter-connected, and cannot be so easily divorced.

Longitudinal stability can be brought into play by itself without affecting lateral, or directional stability, but it has been seen that either of the latter may involve all three.

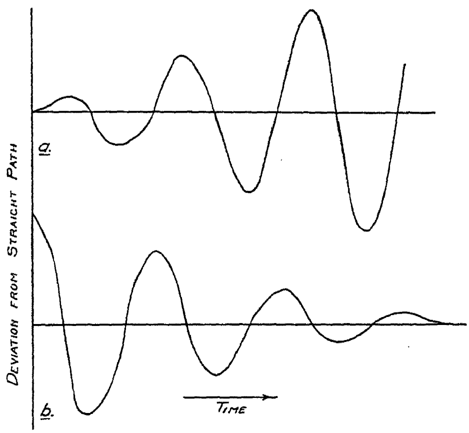


FIG. 65.—FLIGHT PATHS OF UNSTABLE AND STABLE AIRCRAFT

Degree of Stability.

Large commercial aircraft should possess a good degree of inherent stability, so that long flights may be made with the minimum of fatigue to the pilot. Against this, however, it should be remembered that a high degree of stability often results in heavy and slow controls, which may be a danger in itself, besides making hard work for the pilot during manœuvres.

With military aircraft, generally, where powers of rapid manœuvre are essential, inherent stability should be kept to a reasonable minimum, compatible with safety. Past experience is the best guide in all cases, and new types should incorporate stability features similar to those of earlier successful aircraft.

Dynamic Stability.

Stability has been considered as yet only in so far as the effects of any disturbing influence have automatically brought about a righting moment, or a tendency for the aircraft to return to its original state. This is referred to as static stability.

However, it may be that the correcting movement causes the machine to swing in the opposite direction to an even greater extent than the initial deviation, so that an oscillation is set up with increasing amplitude. Such a machine is said to be dynamically unstable, and will not, of its own accord, return to its original condition of steady flight.

For instance, consider the two curves of Fig. 65, which represent the path of flight plotted against time. For the curve (*a*), it is seen that the disturbance, or movement from the straight path, is over-corrected, and conditions become steadily worse, whereas in the case of curve (*b*) the oscillations are gradually damped out until the aircraft returns to steady flight.

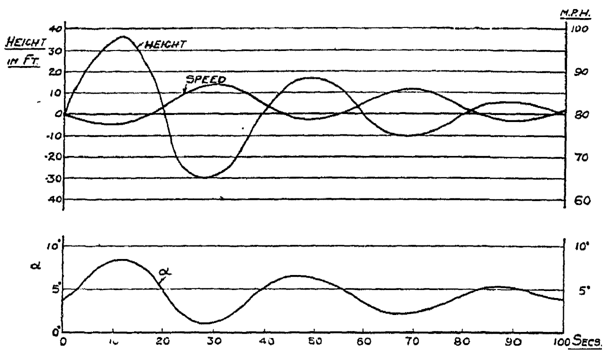


FIG. 66.—CURVES OF HEIGHT, SPEED AND ANGLE OF INCIDENCE FOR RECOVERY FROM PITCHING DISTURBANCE

The behaviour of any one machine can be shown more thoroughly by plotting also curves showing variations in height, speed and angle of attack after some disturbing influence has been exerted. During the test the controls are held fast in their neutral positions.

The curves are for a dynamically stable aircraft, in which the periodic motion gradually subsides after an initial disturbance.

It may be mentioned that a dynamically unstable aeroplane is not necessarily dangerous, since the original disturbance is automatically countered, and allows the pilot time in which to operate his controls.

CHAPTER IX

THE CONTROL SYSTEM AND AUXILIARY DEVICES

Main Control System.

This consists of the control surfaces; ailerons, elevator and rudder; together with the operating mechanism in the pilot's cockpit, and the connecting cables, rods, etc. (Fig. 67). It can be divided into two separate sub-systems, these being the controls actuated by the pilot's control column and the rudder control worked by the rudder bar or pedals.

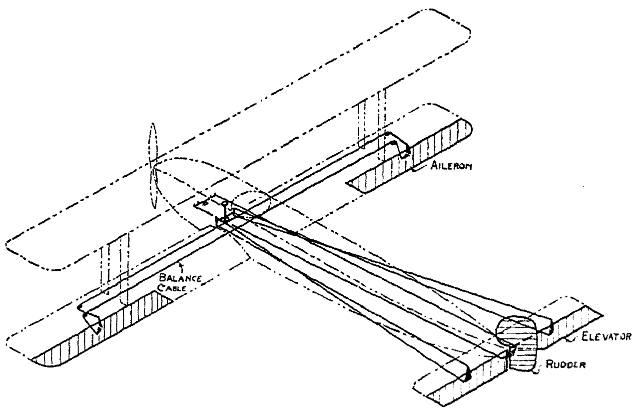


FIG. 67.—MAIN CONTROL SYSTEM

All control surfaces are placed as far as can be arranged from the C_G of the aircraft, in order to provide the greatest turning moments about their respective axes of action. Thus the elevator and rudder are placed at the extreme rear of the fuselage, whilst the ailerons are situated at the outer parts of the main plane for the sake of leverage, and in order to obtain the requisite control power. Shortening of any one arm must be accompanied by enlarged surface area with consequent increased drag.

The purpose of ailerons is to bank the aeroplane, that is to cause rotation about the longitudinal axis.* The rudder is provided to turn the aircraft about the normal, or yawing axis; whilst the elevators control pitching about the lateral axis.

Control Column.

This consists essentially of a vertical column in the pilot's cockpit, hinged near the base, and capable of movement in forward and sideways directions, and in any combination of the two (Fig. 68). Cables connect from the base, or from levers

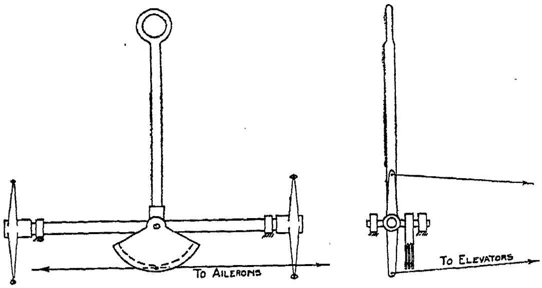


Fig. 68.—Pilot's Control Column

hinged at the base, to the elevators, so that when the control column is moved back the elevators are caused to rise and thus place the aircraft in a climbing attitude. In order to accomplish this the connecting cables are crossed as shown in Fig. 67.

Similarly, cables join the control stick base to the ailerons, so that movement to one side, say to the right, causes the aileron on that side, the right, to rise, and the other to fall. A balance cable connects directly across the ailerons to complete the circuit, or it may also attach to the control column on the opposite side of the hinge, so that failure of the cable would not render both ailerons useless.

In some aircraft the control column is fitted with a hand-wheel for aileron control, in which case rotation of the wheel takes the place of sideways movement of the column. A sprocket wheel, concentric with the hand-wheel, and an endless chain, connecting to a wheel at the base, transmit movements of the control wheel to the aileron cables.

* Chapter VIII.

Rudder Bar or Pedals.

The simplest form of rudder control consists of a horizontal bar at the pilot's feet, hinged at the centre to permit rotation in a horizontal plane. The ends of the rudder bar are connected directly to the control levers on the rudder, so that when the left side of the rudder bar is pushed forward the left-hand rudder lever moves likewise. This brings the trailing-edge of the rudder over to the left, thus setting up a force which moves the tail of the aeroplane to the right and the nose to the left, and so the machine is directed to the left. This control is sometimes claimed to be instinctive, but experience in the instruction of pupils has shown this to be incorrect, since the majority of pupils commence by pushing forward the right end of the bar for a turn to the left, as is usual with bicycles and motor-cycles.

FIG. 69.—RUDDER CONTROL

Pedals are often fitted for the rudder control. In some cases they are connected by universal joints to a rudder bar, whilst in others the pedals are hinged to some rigid part of the floor structure.

Angular Movement of Control Surfaces.

The limiting positions of all control surfaces are generally those at which the maximum force, or moment of force, is available. For the rudder this is about 25°, or even 30°, since in this position the force multiplied by the perpendicular distance of the line of this force from the C_G is greatest. Owing to the decreasing leverage with rotation there is little gain in moment over the last 5°, or so, of rotation, whilst a restriction of rudder movements to within reasonable limits saves some cutting away of the elevators on orthodox designs.

In the case of the elevator, the maximum angle for the tail-plane and elevator combined is roughly 15° to the air-flow, due allowance being made for downwash from the main planes. Above this point stalling conditions are liable to set in.

Elevator angles of about 25° up and down, or a total angular range of 50°, are correct.

Maximum aileron angles may be taken as from 15° to 20°, according to the aileron chord relative to total chord.

It should be remembered that the wing will be already at an angle of, perhaps, 4° in normal flight, and at slow speed when aileron control is generally least effective, the incidence may be near to stalling point, and therefore a large range of aileron angles is not desirable.

This applies, of course, to the down-going aileron and no such consideration affects the raised aileron. Thus it is seen that some improvement becomes possible if the movement can be confined to the up-moving aileron, or, at least, if the down travel is restricted to some extent. At the same time tests have shown that the rolling moment of ailerons drops off suddenly at an angle of 15°.

Differentially Operated Ailerons.

The undesirability of large downward aileron rotation at slow speed, and therefore coarse incidence, has already been referred to. Furthermore, it has been seen* that the purpose of ailerons is to rotate the aircraft about the longitudinal axis. However, during low speed flight, actuation of the ailerons may have just the opposite to the desired effect. For consider an aeroplane flying at a speed just above the minimum and a turn to one side, say the left, is necessary. By means of the control column the pilot attempts to bank the machine to the left and at the same time to yaw the machine in the same direction. The outer (right) aileron has been lowered, but owing to the increased incidence of the right wing, due to the lowered aileron, it is caused to stall, with consequent loss of lift and increased drag, and thus the desired banking does not take place. Moreover, the increased drag of the right wing yaws the aircraft in the direction opposite to that required, so that the intended turn to the left may develop into a disquieting, and probably dangerous, spin to the right.

The obvious solution to the difficulty is an arrangement whereby the action of the ailerons is differential and downward movements are automatically restricted.

The principle of differential aileron control is shown diagrammatically in Fig. 70, from which it is seen that the aileron

* Page 90.

displacements depend on the unequal translational distances moved by two points subjected to equal rotational movement.

Thus the desired banking effect is obtained chiefly by the loss of lift on the (inner) upturned aileron. Also the drag of the upturned aileron is likely to be greater than that of the outer wing, and this, of course, assists the turn by supplying an additional yawing moment in the right direction.

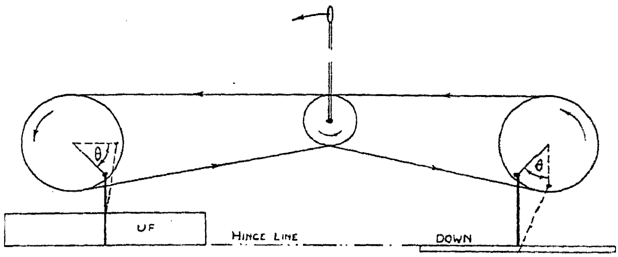


FIG. 70.—DIFFERENTIAL AILERON CONTROL

Differential control is also beneficial at high speed, owing to the reduction of the adverse yawing effect as already explained.

In some cases the down-going aileron is set so that after a little downward displacement it commences to rise again.

Tail Incidence Control.

Another control until recently often fitted to aeroplanes is the tail-plane incidence adjustment control, by means of which the pilot is enabled to :

(*a*) change the loading on the tail-plane to allow for different conditions of flight, and

(*b*) arrange the centre of pressure for the complete aircraft in normal flight to be such that all forces and moments are balanced without the application by the pilot of a continuous force on the control column.*

It is usual to set the tail at a large incidence for assisting in lifting the tail during the take-off,† and at a negative angle for getting the tail down in landing. In normal flight the tail-plane is set in some position, intermediate between these two extremes, depending on the load carried, its disposition, the engine throttle opening (amount of thrust), and on the location of the C_P at different speeds.

* See p. 85. † Page 142.

Actuation of the tail incidence is generally obtained by a chain and sprocket wheel attached to an internally screwed nut, capable of rotation, but not of vertical movement, inside which is a screwed elevating post (Fig. 71 (*a*)). Rotation of the sprocket causes the elevating post to rise or fall in sliding bushes housed in the fuselage structure, and so lifts or depresses the rear spar of the tail-plane, the front tail-plane spar being generally hinged. Any support struts to the tail-plane rear spar should attach also to the elevating post.

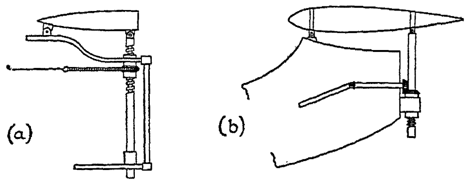


FIG. 71.—TAIL INCIDENCE CONTROL

An alternative method, shown in Fig. 71 (*b*), consists of a bevel wheel drive, the actuating gear being similar to that of an ordinary lifting jack.

Aileron Setting.

Ailerons may be rigged to retain the true aerofoil shape, but are sometimes adjusted with the trailing-edge down (droop), and sometimes in a slightly raised position.

Aileron droop has been incorporated in the past so that the air load during flight would return the ailerons to their neutral position.

Ailerons rigged in either the normal position or slightly up will be raised in flight. This gives a wash-out of incidence over the outer sections of the wing, with a slight differential aileron effect which decreases the tendency to stall and spin during low speed flight.

Partially Balanced Control Surfaces.

The principle underlying the aerodynamic balancing of control surfaces is to reduce the torque on the main strength member, and thus to decrease the force necessary for the pilot to exert. This result is achieved by setting the hinge, about which turning takes place, behind the foremost part of the surface, and so

balancing out to some extent the loads in front of and behind the hinge.

Degree of Balance for Control Surfaces.

The curve XOX′ of Fig. 72 shows the hinge moment plotted against the control surface movement. When part of the surface is situated in front of the hinge smaller hinge moments (or force exerted by pilot) produce similar movements, but, owing to shifting of the C_P, the hinge moments are not proportional to surface movements, and a curve AOA′ is obtained.

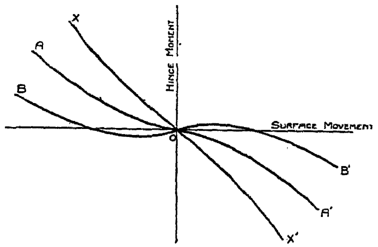


FIG. 72

A greater degree of balance may result in the C_P moving in front of the hinge position so that the hinge moment becomes negative, i.e., the movement, once started, tends to continue without application of additional force. This is known as an unstable arrangement of the control, or overbalancing, and is shown by the curve BOB′. A rough rule for avoiding this is to limit the area forward of the hinge to one-fifth of the total.

It will be appreciated that controls that are perfectly balanced, or nearly so, can be operated with great rapidity by the pilot, since there is little resistance to be overcome. This means that well-balanced controls introduce an element of danger, due to the fact that manœuvres may be carried out too rapidly, causing the application of sudden heavy loads on the main structure. At the same time such rapidity may be most desirable for fighting and acrobatic aircraft.

The following are the usual methods of providing balanced controls :

HORN BALANCE.—The horn balance provides a simple method of aerodynamic balance. The outer end of the surface is extended forward as shown in Fig. 73 (*a*). The modified form,

shown at (*f*), has proved superior to the simple original horn both as regards balance and for prevention of flutter.

Inset Hinges.—In this device the hinges are situated some distance behind the leading-edge, and are supported on arms protruding back from the fixed surface (Fig. 73 (*b*)).

Controllers.—Certain aircraft, notably tailless machines, have the outer parts of the wings rotatable for control purposes (Fig. 73 (*c*)). Here again the turning axis is rearward of the leading-edge.

Elevators built on this principle, with no fixed tail-planes, are termed "pendulum" elevators, but are satisfactory only at very slow speeds.

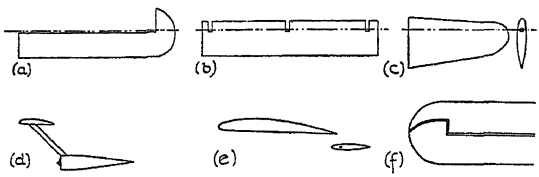


Fig. 73.—Aerodynamic Balance of Control Surfaces

Auxiliary Aerofoil.—Fig. 73 (*d*) shows an auxiliary aerofoil mounted above and in front of the main control surface. It will be seen that this in reality forms part of the control surface and that its forward position gives a balancing effect. The use of this type of balance is generally limited to large aircraft.

Junker Aileron.—This method is similar to (*c*) in that the control surface is a complete aerofoil, separate from the main wing. The aileron being positioned just below and behind the trailing-edge of the main plane produces a slotted effect in the down-turned position with beneficial results at slow speeds.

Fig. 74.—Junker Aileron

Tests carried out with the Junker aileron* showed it to provide roughly 50 per cent. greater rolling moment per unit aileron area than a Frise aileron, and further that the moment is less reduced

* *R. & M.*, 1583, Aug., 1933.

at the stall. Against this, however, the yawing moments for a given rolling moment are somewhat larger except at the stall, where the difference is negligible. It would appear, therefore, that there is little, if anything, to be gained by the use of this device, whilst the additional drag when the aileron is not in use is a further disadvantage.

Apart from the above, however, it was found that almost perfect balance could be obtained with the Junker aileron supported at one-quarter chord, and considerably better balance than is obtainable with the Frise balance hinged at 27·5 per cent. of the chord back from the aileron leading-edge. This suggests that the device might be suitable for very large aeroplanes. An aileron chord of one-fifth main-plane chord was used and of five relative positions tried out, the best is as shown in Fig. 74.

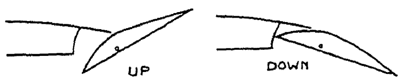


FIG. 75.—FRISE BALANCED AILERON

Frise Balanced Ailerons.

The Frise aileron is a form of inset-hinge control, so arranged to give close balance (Fig. 75). Difficulty is experienced in balancing by setting back the hinges, owing to the shielding effect of the main wing, which gives very different proportion of balance when the aileron leading-edge is exposed and unexposed. The Frise aileron was introduced to overcome this defect, and is obtained by lowering the aileron leading-edge to the lower main wing surface, so that the nose protrudes below the main wing when the aileron is in the " up " position.

The force required to depress the down-going aileron, in the early stages at least, is in excess of that required to raise the other. The underlying principle of the Frise aileron is the almost immediate exposure of the leading-edge of the up-going aileron, which thereby gives an over-balance effect to the one aileron, and thus assists in depressing the other. By this means it is possible to obtain a very close balance, with improved rolling qualities and a decrease of the adverse yawing effects.

There is a danger of overbalancing due to both ailerons being rigged slightly up, which is accentuated by cable slack allowing the air loads to raise the ailerons. This may be overcome by so shaping the ailerons that the nose is slightly above the main plane lower surface in the neutral position, as shown dotted in Fig. 75.

The drag of the Frise arrangement is considerably reduced by the addition of a " curtain " over the upper side of the gap.

A modification of the Frise aileron has been obtained by giving an upward twist to the leading-edge, which results in a graduated balance effect.* This is an improvement in so far as control is concerned, but suffers from the disadvantages of loss of lift at large aileron angles, due to the protrusion of the aileron leading-edge above the top main surface, and the fact that curtaining is not possible, over some portion at least, where this occurs.

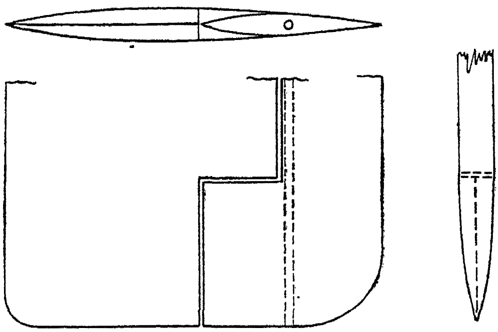


FIG. 76.—GRADUATED BALANCE

Graduated Balance.

The graduated balance of control surfaces is an arrangement whereby the balancing effect increases with the angular displacement of the surface. Fig. 76 shows a method of effecting this result. The control surface is horn-balanced, with the leading-edge of the horn portion tapered down, whereas the rear portion of the fixed surface in front increases in depth away from the tip. Slight movement of the control exposes a small portion only of the horn balance, whilst the remainder is shielded.

An obvious disadvantage of this arrangement is the upsetting of the air-flow caused by the discontinuity of the aerofoil surface.

* *R. & M.*, No. 1587.

Servo Control.

The servo control consists essentially of a small auxiliary surface situated at a relatively large distance, generally to the rear, from the hinge position of the main control surface to which it is attached (Fig. 77 (*a*)). The control cables are attached directly to the servo, which is free to rotate on its own hinges, so that a small force induced on it by rotation in one direction causes movement of the main control surface in the opposite direction.

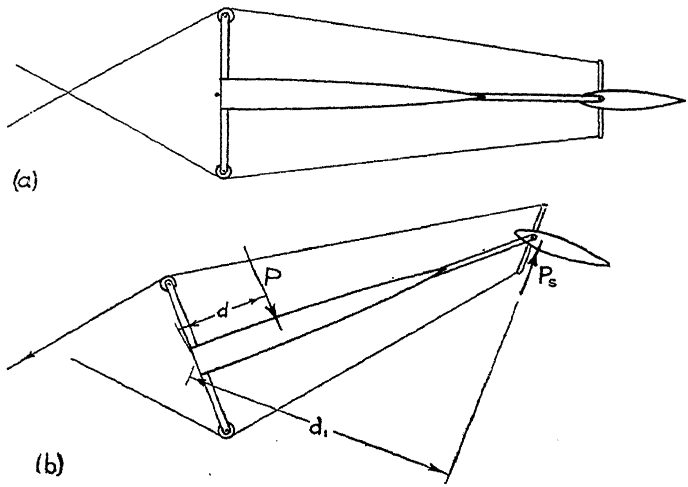


FIG. 77.—SERVO CONTROL

The main force $P = \frac{P_S a_1}{d}$, and the nett turning moment for the aircraft is the difference between the moments of P and P_S about the C_G (Fig. 77 (*b*)).

The position of the servo to produce a force P_S is affected by the " downwash " from the main surface.

It will be noticed that the control cables must be crossed for this method of control, and that the crossing may take place either between the two hinge positions, or in front of the main hinges. Arms supporting pulleys may be fitted to the main spar for holding the cables clear of the moving surfaces.

Balance Tab.

The balance tab, whether external or inset, is similar in action to the servo control, but with a different method of actuation.

The flap is hinged to the main surface and is connected to a fixed part of the aircraft by a push-rod as shown in Fig. 78. Movement of the main surface in one direction causes the flap to rotate in the opposite direction, and thus to set up a force at the trailing-edge which assists the main movement.

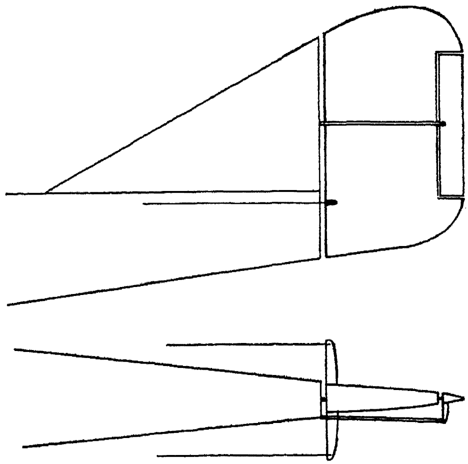


FIG. 78.—BALANCE TAB

Trimming Tab.

The trimming tab has been introduced to supersede the tail adjusting gear and the rudder bias strip. It consists of a small tab situated at the rear of the main control surface, and is provided with a control mechanism, separate from the main control and actuated from the pilot's cockpit. By arranging for the tab control cables to pass through the hinge position of the main control surface, the relative position of the tab and main control are unaffected by movement of the latter. The advantages of the trimming tab over the adjustable incidence tail-plane are a saving in weight, a simplification of the tail construction resulting in greater strength and rigidity, together with lightness and quickness of operation. Against these must be offset a probable small increase in the tail drag and decreased efficiency of the control surface, due to initial loading caused by the tab setting, which may necessitate the use of a slightly larger control surface.

Narrow chord flaps are almost equally effective as large chord flaps at small elevator angles but fall off at coarse angles,

owing to their being situated in the region of turbulent wake, but since balance is required chiefly at high speed flight where coarse use of the elevator is not required, this loss of effectiveness is not important. A trimmer area 5 per cent. of the tail area, situated along practically the whole elevator span,* appears to give most satisfactory results. The angular movement of such tabs is three times that of an adjustable incidence tail-plane for similar results if the elevators are unbalanced, becoming less with elevator balance.

Tests have been made with the trimmer mounted external to the main control surface,† but the results were inferior to those for the inset flap.

The use of trimming tabs on rudders of multi-engined aeroplanes has enabled twin-engined aircraft to be produced with a single vertical tail, having satisfactory flight characteristics with one engine only effective.

The Vee Tail.

The normal tail unit of cruciform shape, with horizontal and vertical tail surfaces, is not entirely satisfactory. When elevators and rudder are actuated together there is mutual interference due to the upsetting of the positive and negative pressure regions. In certain manœuvres, notably the spin, considerable blanketing takes place. In both cases there is a diminution of efficacy of the controls.

An attempt has been made to overcome these undesirable features by means of a vee tail,‡ in which the two halves of the tail-plane are set at a considerable dihedral angle, vertical surfaces being entirely dispensed with. Actually this is a closer imitation of Nature's aircraft, but still falls short of the ideal as exemplified by our feathered masters, the birds.

Elevators are hinged to the stabilising surfaces and may be worked together for longitudinal control, or differentially for turning. This differential action provides both a yawing moment and a rolling moment, the latter acting against the ailerons if the tail dihedral is positive, and with the ailerons (favourable) if the dihedral is negative.

* " Notes on Trimming Flaps for Tailplanes." Report No. B.A. 1185 (Revised), Feb., 1935.

† " Control Surface Flaps for Trim and Balance." *The Aircraft Engineer*, supplement to *Flight*, Feb. 28th, 1935.

‡ " The Rudlicki Vee Tail," *Aircraft Engineering*, March, 1932.

When used as elevators, horizontal force components are set up, which, although cancelling out so far as yawing of the aircraft is concerned, add both to the structural loads and to drag, and of course reduce the effectiveness of the tail.

Tests in the wind tunnel and in actual flight have proved the vee tail to be at least as efficient as the conventional tail, with approximately a 25 per cent. decrease in total area. Thus a saving of both weight and drag should be possible with this tail arrangement.

Mass Balancing of Control Surfaces.

The balancing of control surfaces dealt with earlier in this chapter* deals only with the aerodynamic loads falling on them and is done to lighten the forces required to be exerted by the pilot.

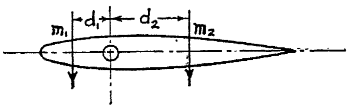


FIG. 79.—STATIC BALANCE OF CONTROL SURFACE

Control surfaces are often statically balanced by balancing out the moments of all masses before and behind the hinge position. Thus in Fig. 79, if $m_1d_1 = m_2d_2$ the control surface is statically balanced, and in order to achieve this, extra weights are sometimes supported forward on projecting arms.

Of recent years, the higher speeds attained with aeroplanes has caused the problem of flutter to be given much more careful consideration, and as a preventive of flutter, mass balancing has been introduced, particularly in respect of ailerons.

It may be said that a statically balanced control surface is approximately mass balanced also, but the latter condition is dependent also on the distance of the mass centre from the root of the wing. Thus an aileron is said to be mass balanced if the value of $M\,x\,y$ is *small and negative* (Fig. 80), where M = total mass, and x and y are distances of the centre of mass from the aileron hinge line and wing root. This means that the aileron C_G should be close to but slightly in front of the hinge line, and that the greater y is made the smaller x need be.

* Page 106 *et seq.*

The desired result can be achieved by loading the aileron leading-edge, or by the addition of a bob-weight extending forwards above, below, or inside, the main wing, or by a combination of both. By placing the bob-weight well outboard, it is possible to obtain the necessary balance with a relatively small additional weight.

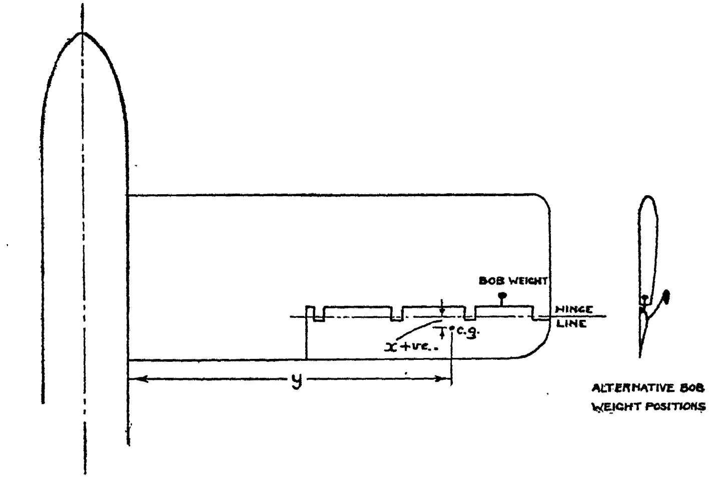


FIG. 80.—AILERON MASS BALANCE

CHAPTER X

THE SLOTTED WING AND VARIABLE LIFT ARRANGEMENTS

UNDER this chapter heading are dealt with the many devices for assisting the taking-off and landing of aircraft. One of these devices, namely, the slotted wing, has functions other than those just specified, and is employed in a number of ways, for which reasons it is given special attention here and is fully dealt with in its various forms before going on to a consideration of those devices which are of use only for augmenting the speed range of aeroplanes. At the conclusion of the chapter the suitability of all arrangements for the duties last mentioned is discussed.

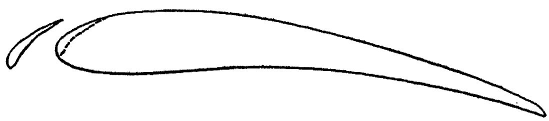


FIG. 81.—THE SLOTTED WING

The Slotted Wing.

The slotted wing, Fig. 81, is a device for increasing the aerodynamic efficiency of an aerofoil at large angles of attack and therefore is generally associated with the slower speeds of flight. In brief, the presence of the slot delays the breakdown of smooth air-flow conditions until a larger angle of incidence is reached, and thus enables higher lift coefficients to be attained. The benefits derived are twofold.

(*a*) Slower speed becomes possible without alteration of wing area. In other words the requisite value of lift can be obtained at slower speeds, and

(*b*) The smooth flow conditions, at angles above the normal stalling angle for the aerofoil, enable the control surfaces forming part of the wing to retain their powers of control over the increased range.

The results are achieved by the utilisation of a small auxiliary aerofoil, or slat, situated just forward of the leading-edge, and which is generally held back to form part of the leading-edge at the smaller angles of incidence.

Explanation of Action.

It will be noticed that the slotted wing is in reality a combination of two wings, and that the forward aerofoil is situated in the upwash of the main wing. The air-flow can be traced over the leading aerofoil and continued on over the rear wing, where it is found that the angle made by the air-flow meeting the main wing is less than the apparent incidence. In other words, the main wing is in the region of down-wash from the front wing, and the air, having been deflected downwards to some extent already, finds less difficulty in following the upper contour (: points A and A′ in Fig. 82).

FIG. 82.—ACTION OF SLOTTED WING

It may be mentioned, in passing, that, apart from other considerations, the open slotted wing gives an increased wing area and the lifting force likewise must be augmented, but this in itself is not a sufficient explanation of the greater lift obtained by means of this device.

The slat chord is generally set at an angle of roughly $-40°$ to the main chord, though the effective incidence is relatively

small on account of the pronounced upwash in this region. The shape of the channel may be varied within fairly wide limits, but the rear opening, i.e., at the trailing-edge of the slat, is generally arranged to be between 2 and 3 per cent. of the main chord. The front opening is about two to three times that at the rear and varies according to the rear gap. This has the beneficial effect of speeding up the air-flow over the upper surface in the region of the boundary layer and thus delaying the advent of the break-away.

Automatic Opening and Closing.

If we refer back to Fig. 24,* giving the pressure distributions over a wing for different angles of incidence, we are able to find the nature of the force acting over the forward part, constituting the auxiliary aerofoil, in the closed condition. At small angles the pressure over this part is mainly positive, i.e., is greater than atmospheric and tends to hold the winglet down to the main wing, but at larger angles the negative pressure region extends further towards the leading-edge, and thus a pressure difference is set up between the top and bottom surface of the auxiliary which tends to lift it (Fig. 83 (*b*)).

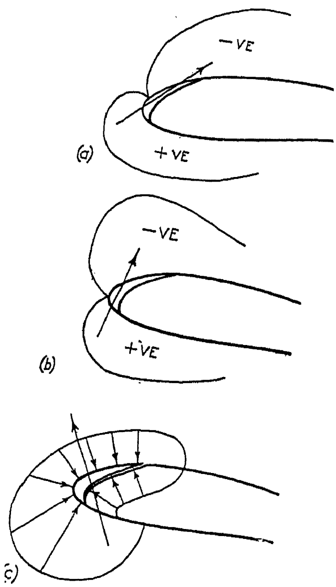


FIG. 83
SLOT PRESSURE DIAGRAMS

Between the slat and main aerofoil in the closed position there is generally a small gap. The pressure within the gap is roughly the same as that obtaining at the entrance, that is, above atmospheric if the opening is at the bottom, and below atmospheric if at the top. This allows for a certain amount of adjustment of the pressure on the underside of the slat and hence the timing of the slot opening can be controlled to some extent by suitably arranging the gap setting.

The complete pressure diagram is illustrated at (*c*), which shows the total resultant force acting on the slat to be forward

* Page 36.

of the vertical. The slat is thus constrained to move forwards; its path and relative position being controlled by suitable link mechanism.

The next diagram, Fig. 84, shows for one slot arrangement the magnitude and direction of the slat force, in the open position, for various incidences of the main wing. Below 8° the tendency is for the slat to close, but above 8° the forces are directed well forward.

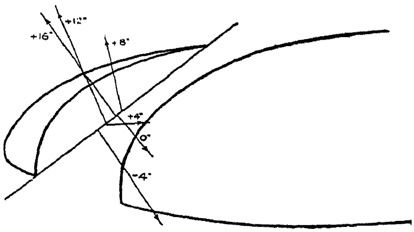


FIG. 84.—SLAT FORCES

For any particular wing section, opening can be arranged to take place at a desired angle of incidence by proper shaping of the slat, arrangement of the link mechanism, and adjustment of the gap as previously described. Pressure diagrams over a range of incidences are necessary for this purpose. A spring is generally incorporated to prevent flutter of the slat at the critical point of balance.

Use of Slotted Wing in Practice.

The introduction of the slotted wing enables the lift coefficient to continue to increase up to a value of about 1·7 at an angle of roughly 22° (see Fig. 87), a value which is sensibly constant for all aerofoil shapes, from which it is obvious that the benefits of a slot as regards lift are greatest in the case of thin, or "low lift," sections. The increased lift is accompanied by a reduction of drag coefficient values as compared with the drag of the plain aerofoil at corresponding angles of incidence. Unfortunately the closed slotted wing generally offers greater drag than a continuous aerofoil, whilst any external arms, or supports, required for the link mechanism also add to the total resistance.

The most usual position for slots is over the outer section of the wing span, in order that the ailerons may retain their effectiveness at large angles of attack and thus eliminate the

dangerous tendency of involuntary spinning from the stalled attitude. It will be apparent that a slotted top wing, of a biplane, particularly where positive stagger is incorporated, may assist in maintaining smooth air-flow conditions over the lower wing also.

Forward Fixed Auxiliary Aerofoil.

Slotted wings have been tried out with the slot permanently open, but the high C_D values at small angles of incidence are very detrimental to high speed performance. Tests have been carried out in America on scale models in the wind tunnel, and on full-size aircraft in flight, with a small assisting aerofoil placed ahead of the main wing (Fig. 85). In the wind tunnel tests* the auxiliary was a highly cambered aerofoil of medium thickness having a chord 14·5 per cent. of the main wing and was tested in no less than 141 different positions ahead of, above, and behind the main wing leading-edge.

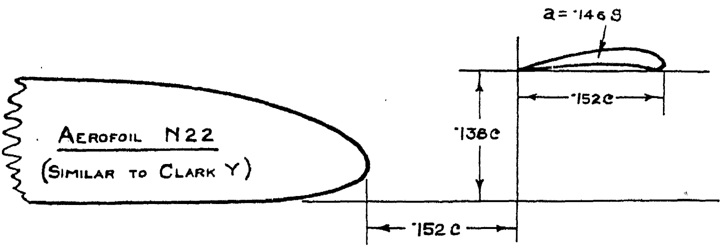


FIG. 85.—FIXED AUXILIARY AEROFOIL

Positions were found in which for a 30° negative setting of the auxiliary angle $C_{L\,max}$ was as high as 1·81, or 40 per cent. greater than for Clark Y alone, but the corresponding value of

was only optimum position was found to be
$_{D\,min}$
with the trailing-edge of the auxiliary wing 0·15 c ahead of, and 0·12 c above the main leading-edge, and with the auxiliary wing chord either parallel to, or at + 2·5° to the main chord. In such position $C_{L\,max}$ was found to be 1·705, an increase of 32 per cent. over the plain wing, and the $\frac{C_{L\,max}}{C_{D\,min}}$ value was 104·5, these values being not greatly inferior to those for the

* "Wind Tunnel Tests of a Clark Y Wing with a Narrow Auxiliary Aerofoil in Different Positions," *N.A.C.A. Report*, No. 428.

ordinary slotted aerofoil, whilst $C_{D\ min}$ for the optimum position referred to above was 0·0187, a vast improvement over minimum drag for a fixed slot.

The effect of adding an auxiliary wing over the whole span of a conventional monoplane, assuming no additional weight, is said to change a speed range of 50–115 m.p.h. to 41–112·5 m.p.h., or compared with a plain wing of the same total area the former figures would be 47–113, which shows a decrease in V_{min} of about 13 per cent. for a loss of less than 0·5 per cent. in the top speed.

The full-scale flight tests* were carried out to obtain confirmation of the tunnel tests. The auxiliary wing arrangement is shown in Fig. 85, for which the following results were obtained.

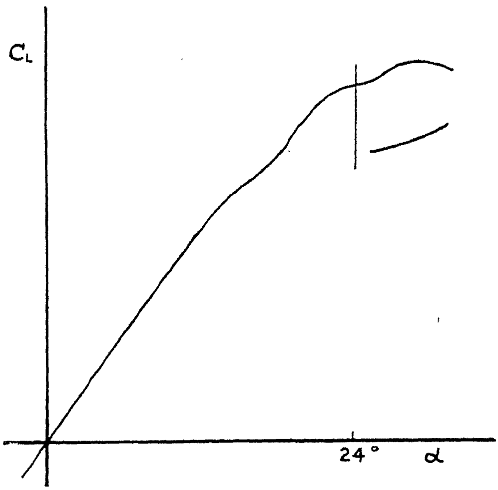


FIG. 86.—C_L CURVE FOR AUXILIARY AEROFOIL COMBINATION

$C_{L\ max}$, based on the original wing area, showed an increase from 1·35 to 1·96, $C_{D\ min}$ increased from 0·05 to 0·052, $\frac{C_{L\ max}}{C_{D\ min}}$ for the aeroplane increased from 27 to 39, whilst $\frac{L}{D}$ (max) for the whole aircraft remained unaltered at 9·3. The landing speed was lowered by 9 m.p.h. and increased the flight range by some 10 per cent. Against these improvements however, the take-off, climb, and ceiling were rendered worse.

* "Flight Tests to Determine the Effect of a Fixed Auxiliary Aerofoil on the Lift and Drag of a Parasol Monoplane," *N.A.C.A.* Tech. Note No. 440.

A curious feature, noticed also with other similar wing arrangements, is the double peak of the C_L curve, shown in Fig. 86. The maximum value of the lift coefficient has been given as 1·96, which is the value at an angle of incidence of 24°. Above this angle there is a further increase, but, owing to the instability caused by sudden transition from the higher value to the corresponding lower value, it is considered unsafe to exceed this angle in flight. It may be remarked that an angle of attack approaching 24° is generally considered too high for practical use to-day, and that the various forms of flap gear offer greater promise for future development both for this reason and because of their lack of interference, or even improvement, during take-off and climb, but the use of fixed auxiliary aerofoils, forward of the aileron sections of a wing, as a device for improving lateral stability, particularly at low speeds, is well worth the serious consideration of those who place the quality of safety high in the list of desirable attributes of an aircraft's performance.

Pilot Planes.

A variation of the slotted wing is provided by the combination of a main aerofoil with an auxiliary, or pilot plane, which is permanently supported just forward of the leading-edge. This latter is pivoted at, or near, its leading-edge, and is free to rotate at the lower angles of incidence of the main plane. As the incidence increases, the rear edge of the pilot plane becomes lifted in the upwash forward of the main aerofoil, but at some degrees prior to the normal stalling angle the pilot plane comes up against a stop and thus becomes a lifting surface. Its position now relative to the main wing is similar to that of a normal slat in the open position and as such tends to increase the $C_{L\,max}$ value and delay stalled conditions.

This arrangement gives only slightly better results than the fixed forward aerofoil, so that the additional mechanical complications are scarcely warranted.

The Slotted Aileron and Wing Flap.

Greatly increased control power can be obtained by the incorporation of a slot between the main wing and hinged control surface. Wing flaps, for varying the camber, dealt with later in this chapter, are also sometimes slotted. The slots between

the main surface and the flaps are similar to the forward slots previously described, as also is the aerodynamic principle involved.

The first air-flow diagram of Fig. 82* shows how the flow at large angles becomes turbulent over the rear part of the aerofoil, and obviously this would be accentuated by a down-turned aileron. This is obviated, to a large extent at least, by the presence of the slot. The resulting pressure diagram is illustrated in the next figure.

Still greater lift is obtained by the combination of a slotted main aerofoil together with slotted flaps. It is recommended that both ailerons and elevators should incorporate the slot arrangement when in the down position.

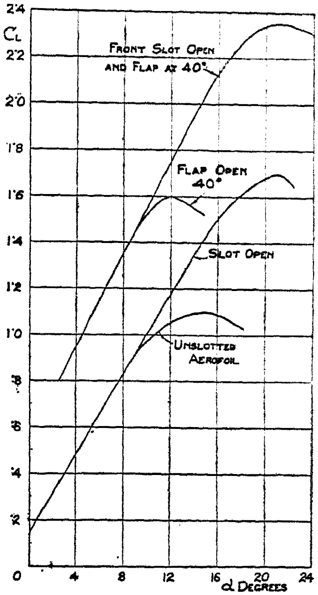


Fig. 87. — Lift Characteristics of Aerofoil with Slot and Flap

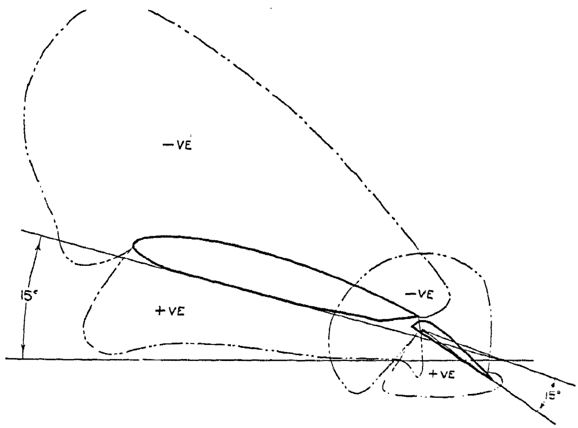


Fig. 88.—Pressure Diagram for Aerofoil with Slotted Aileron

The Multi-slot.

This is merely a development of the single slot. Each winglet

* Page 117.

imparts a small relative downward motion to the air-flow, the effect being cumulative. It has been shown,* however, that the downwash reaches its maximum inclination at the fourth slat of a series, the straightening effect of the free air-stream beyond the wing system being sufficient to damp out further tendency to increased downward deflection. High lift coefficient values are obtainable by this means (probably exceeding $C_L = 4{\cdot}0$), but the high drag shown to be present in experiments so far conducted, together with the inherent mechanical difficulties, have prevented its practical application. It is, however, a device freely adopted in bird flight, both for wings and tails, and is likely to find a use in aircraft at some future date. Experiments have been carried out to ascertain the effect of "slotting the slot," i.e., in which a very small slot has been fixed over the front of the larger slot, although this is really only a special form of the multi-slot.

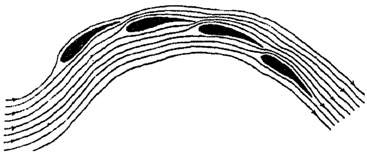


FIG. 89.—THE MULTI-SLOT

The Interceptor.

Under certain conditions it is undesirable for the effects of the open slot to be present. For example, if a turn is to be made to the left at a coarse angle of attack, such that the slots on both wings are open, then loss of lift on the inner, or left, wing, accompanied by increased drag, is required, and this result is obtained with slotted wings by means of a plate or interceptor, hinged to the upper surface of the main plane, just behind the trailing-edge of the slat, and which may be raised at the pilot's will (Fig. 90).

FIG. 90.—THE INTERCEPTOR

Normally the interceptor lies flush with the main surface,

* *R. & M.*, 639.

but in the raised position it restricts the slot passage very considerably, and thus causes turbulent motion of the air-flow to be set up, with consequent reduced lift and additional drag. In other words, both rolling and yawing moments of the desired nature are provided. The interceptor may be inter-connected with the aileron so that upward movement of the aileron is accompanied by raising of the interceptor.

Townend Ring and N.A.C.A. Cowling.*

The main purpose of the small extra aerofoil of the slotted wing arrangement has been seen to deflect the air-flow downwards over the main wing and so prevent burbling. It is evident that the same principle may be adopted in other places where it is desirable to change the direction of flow. The Townend ring is fitted to the front of a fuselage, or nacelle, with a blunt nose (radial engine) for this purpose. The ring is of aerofoil shape in section and is placed where the air-flow is being deflected outward, away from the fuselage, so as to assist in bringing it back, more or less, to the original direction of flow (see Fig. 91).

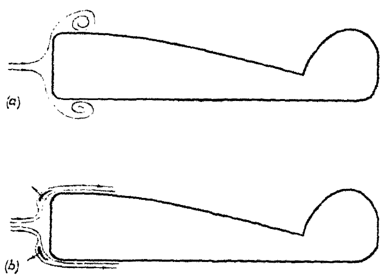


FIG. 91.—TOWNEND RING

It will be noticed that the resultant air force on the ring acts in a forward direction, the total radial components cancelling out of course, though this in reality merely represents a part of the drag loss, the balance being due to the relatively smooth flow established over the fuselage. The drag decrease gained by the aid of the Townend ring is roughly about half that of the fuselage and engine, or perhaps two-thirds of the engine drag.

The N.A.C.A. cowling, developed about the same time, is similar to the ring just described, but extends further to the rear so as to cover the front of the fuselage and by leaving only a small annular gap is more truly analogous with the slotted wing. This annulus is given an area of about one-half the front opening in order to gain the benefit conferred by the venturi effect.

The N.A.C.A. cowling gives results similar to the ring in most

* See also p. 197, *et seq.*

instances, and appears superior for bodies in which the greatest diameter is situated some distance back from the nose.

Variable-Lift Devices.

The advent of the aerodynamically clean design of modern aircraft, with its consequent fine angle of glide, together with the ever-increasing wing loading, have meant that the length of landing run has been greatly increased during the past few years. This factor has made it necessary to devise some means of decreasing the forward speed (increased $C_{L\ max}$) and increasing the angle of glide (increased C_D).

The desirable requirements of any such device have been enumerated by Gates* as follows:

1. Increase in $C_{L\ max}$ by at least 50 per cent.
2. Ratio of glide for landing approach to be at least 1 in 6 at a speed 20 per cent. greater than stalling speed.
3. No increase of incidence of the basic section should accompany the use of the device.
4. The device should be controllable by the pilot, reasonably light, light to operate and capable of use in various positions from the closed to fully open.
5. Extra drag in closed position should be low.
6. Its use should cause little change in longitudinal trim, control and stability.
7. The use of normal ailerons should be possible.

Many methods of increasing the speed range of aircraft have been made, with some considerable measure of success, and such are generally referred to as "variable-lift" devices, although, strictly speaking, variable-lift is a misnomer brought about, no doubt, by the erroneous belief often held that the lift of an aeroplane changes with change of speed.

It should be remembered that during all normal horizontal flight, apart from the accelerations accompanying manœuvres, the lift must be sensibly equal to the weight, and since for all practical purposes it may be said that the weight is constant, so also must the lift be.

The main object underlying all these arrangements is to increase the wing lift coefficient beyond the normal $C_{L\ max}$ and so

* "Trailing-Edge Flaps in Relation to Take-off and Landing of Landplanes," by S. G. Gates, M.A., *R. & M.*, 1707.

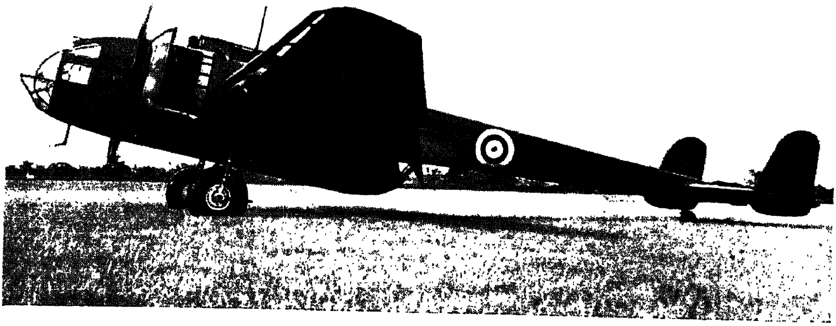


FIG. 92.—HANDLEY PAGE "HAMPDEN"
NOTE—SLOTS, FLAPS, ADJUSTABLE ENGINE COWL FLAPS AND UNDER-TAIL GUN BALCONY
("*Flight*" *Photo*)

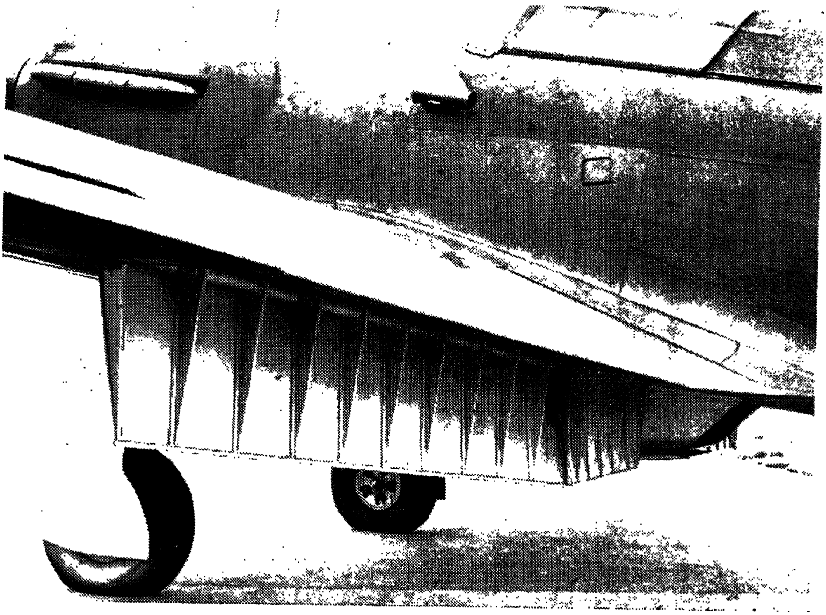


FIG. 93
SPLIT FLAPS, HYDRAULICALLY OPERATED—HAWKER "HURRICANE"
(*Reproduced by courtesy of "Flight"*)

[*Facing page* 126

obtain a greater speed range, either by rendering possible lower landing speeds, or more generally by effecting a reduction in wing area, without altering the minimum speed, with consequent increased top speed. Alternatively the speed range may be augmented by means of an adjustment of the effective wing area.

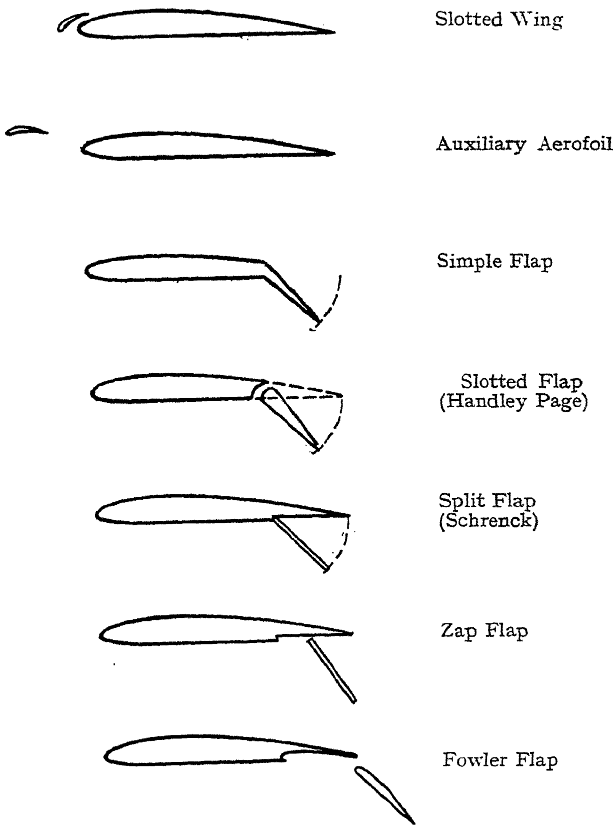


FIG. 94.—VARIABLE LIFT DEVICES

With most variable-lift devices the increased lift coefficients are accompanied by high values of the relative drag coefficients, and this is made use of for steepening the angle of glide for landing approaches. Other devices give improved lift values with little increase of drag and may be employed to advantage in assisting the take-off of aeroplanes with high wing loading.

Variable Wing Area.

Attempts to change the wing area during flight have taken the form of variable span and variable chord arrangements, in each of which sliding extensions have been incorporated, which are operable by the pilot. Some success has been claimed, but as the tests have not progressed beyond the experimental stage it is too early yet to give a definite ruling on their respective merits.

It would, however, certainly appear that the aerodynamic advantages accruing from increased span would be greater than from increased chord, though the reverse would be the case whilst alighting. Apart from the extra weight and complications of the actuating mechanism for any such device, it is obvious that there must be an increase in structure weight and, perhaps of more importance, some redistribution of loading is unavoidable, although here it will be appreciated that the variable chord arrangement, when open at slower speeds, would tend to keep the centre of pressure further back from the leading-edge, or more coincident with the top speed position, and could be arranged to reduce the total C_P travel.

Another method of increasing the surface area that has received attention is by forming the rear under-surface of the wing into a separate auxiliary aerofoil, of from 25 to 50 per cent. of the main plane chord, that may be slid down and back when desired. As many as two and three of these small auxiliaries have been tried.

Variable Camber.

In all natural flight the wings possess a high degree of flexibility, so that in high speed flight, when the centre of pressure tends to retire from the leading-edge, the camber decreases with a corresponding reduction of the drag coefficient value and thus a greater speed range becomes possible. Or conversely, as speed is decreased the camber increases, which thus gives higher lift coefficient values, and permits the attainment of slower minimum speeds. A further advantage must be the decrease in wing torsion due to the restriction of the rearward C_P movement.

In mechanical flight these benefits have been sought in two ways. In the first the portion of the ribs behind the spar has been made flexible in a fairly close imitation of the bird's wing,

whilst the second, and more popular at the present time, is to provide a hinged flap over the rear portion of the wing, which may or may not be left to the pilot to operate.

Without doubt further development of the variable camber wing will take place in the near future, since this offers definite increased efficiency at the cost of very little additional weight.

Trailing-Edge Flap (Simple).

This device in its unslotted form at least may be regarded as a variable camber wing.

The effect of depressing the rear portion of a wing is to "bank up" the air below, with consequent decreased velocity and increased pressure on the underside. Also a dead air region is formed above the flap which assists in keeping the upper flow from breaking away and thus increases lift by some 40 to 60 per cent. beyond the normal $C_{L\ max}$. The lift increment is greatest, as might be expected, with those sections having small mean-line camber and therefore low normal $C_{L\ max}$.

Best results are obtainable with a flap chord of about 0·3 c, but the loss for a reduction to 0·2 c is in the neighbourhood of 5 per cent. only. A short chord flap, say, of 0·1 c, gives highest lift with a 75° flap angle whilst for a chord of 0·4 c a flap angle of 30° only is required. Flap angles seldom exceed 40° in practice.

The high values of lift due to flaps are met with at angles of incidence rather below that of $C_{L\ max}$ for the initial section. Thus the aeroplane is not caused to adopt a coarse attitude during the approach to land and an extra deep landing chassis is not required.

It may be remarked here that autorotation* is brought about by downward movement of the outer aileron when the main wing is already at a high angle of incidence, so that the new attitude is beyond the critical position for the flapped aerofoil and loss of lift results.

The gain in lift due to partial span flaps when situated at the inner part of a wing is greater than the proportion of flapped span. For example, flaps over the centre half-span give roughly 60 per cent. of the gain for full span flaps. This arrangement leaves the outer half of each semi-span available for ailerons.

* See p. 148.

Trailing-edge flaps are sometimes of assistance during take-off on account of the lower values of $\frac{C_D}{{C_L}^{1\cdot5}}$ known as the power coefficient, with flaps depressed through an angle of about 5°.

Much of the above applies also to flaps other than the simple trailing-edge, descriptions of which follow.

The slotted trailing-edge flap has already been dealt with. Here lift is augmented by means both of the flap and the slot effect, giving a 15 per cent. greater $C_{L\ max}$ and a lower drag than in the case of the simple flap.

Split Trailing-Edge Flap (Schrenck).

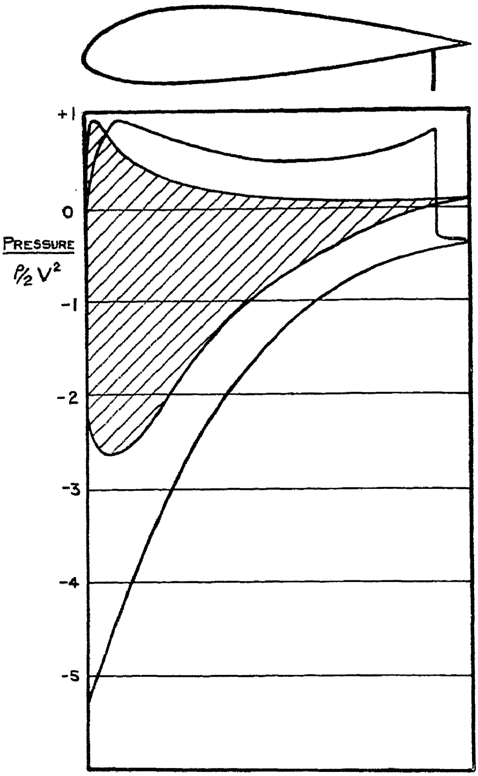


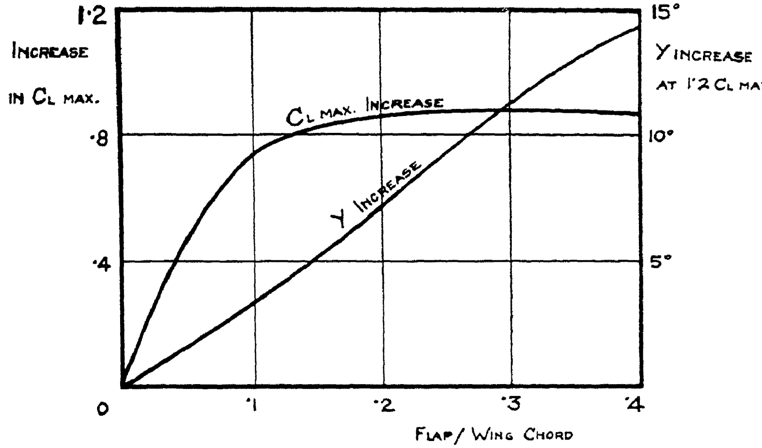
FIG. 95.—PRESSURE DISTRIBUTION OVER AEROFOIL WITH SPLIT FLAP

In the Schrenck flap arrangement the under surface of the rear portion of the aerofoil is hinged to fold down, leaving the upper surface unaltered. The aerodynamic effect of the open flap is seen from Fig. 7 to be a lowering of pressure in the dead air region within the "jaw," which helps to maintain a down-flow above the wing and thus prevent the breakaway, with delayed stalling.

Fig. 95 shows the pressure plotted over a wing without a flap, with the pressure diagram for the same wing when fitted with a Schrenck flap of one-tenth chord, from which it is seen that the increased lift is due chiefly to the lowered pressure over the top surface.

Experiments* have been undertaken to ascertain the effects of varying the flap chord, the flap angle, and the flap position relative both to the main plane chord and span.

The effects of variation of the flap chord are shown in Fig. 96, from which it is seen that $C_{L\ max}$ increases with flap chord up to 20 per cent. of the main chord, where a 70 per cent. lift increase is experienced, but that there is little further gain in lift. The angle of glide, γ, appears to increase fairly uniformly with the flap chord, but in view of the heavy operating loads and weight of flap gear it is doubtful whether a flap chord in excess of one-fifth of the total chord is warranted except in extreme cases where a very coarse gliding angle is essential.



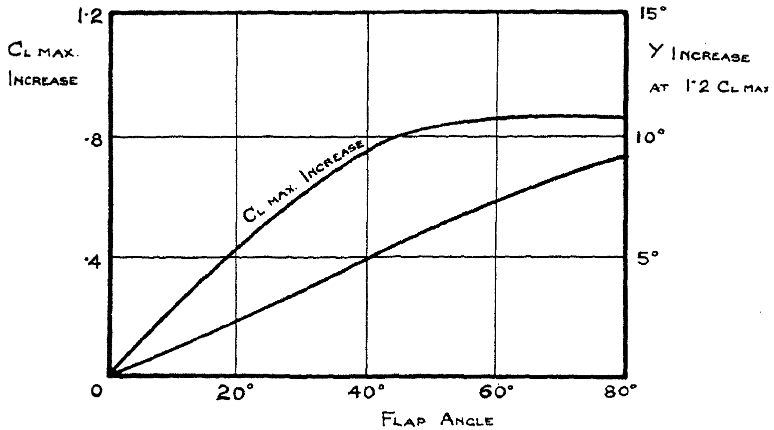
(FLAP ANGLE 60°)
FIG. 96.—EFFECT OF FLAP CHORD VARIATION

The next figure (Fig. 97) indicates that so far as $C_{L\ max}$ is concerned little advantage is gained by opening the flap to a greater angle than 50°, or perhaps 60°, and the latter is generally taken as the limiting position, though as with flap chord variation the angle of glide continues to show increase with greater opening up to 100°. With flap chords of less than 20 per cent. an angle in excess of 60° appears to give beneficial results and an angle of 90° is sometimes used in practice.

Tests made by the Author on the effect of hinging the flap at various positions along the main-plane chord show the gain

* "Split Flaps and Other Devices for Facilitating Landing," *R. & M.*, No. 1659.

in $C_{L\ max}$ to increase as the hinge position is moved back (Fig. 98). This result has been borne out by other tests. It would be expected that forward movement of the flap would adversely affect the pressure distribution over the top and bottom surfaces of the aerofoil, whilst of course a flap trailing-edge to the rear of the main trailing-edge has the additional effect of increasing the total chord of aerofoil and flap. The significance of these factors will be apparent during the consideration of the Zap flap in a later paragraph.



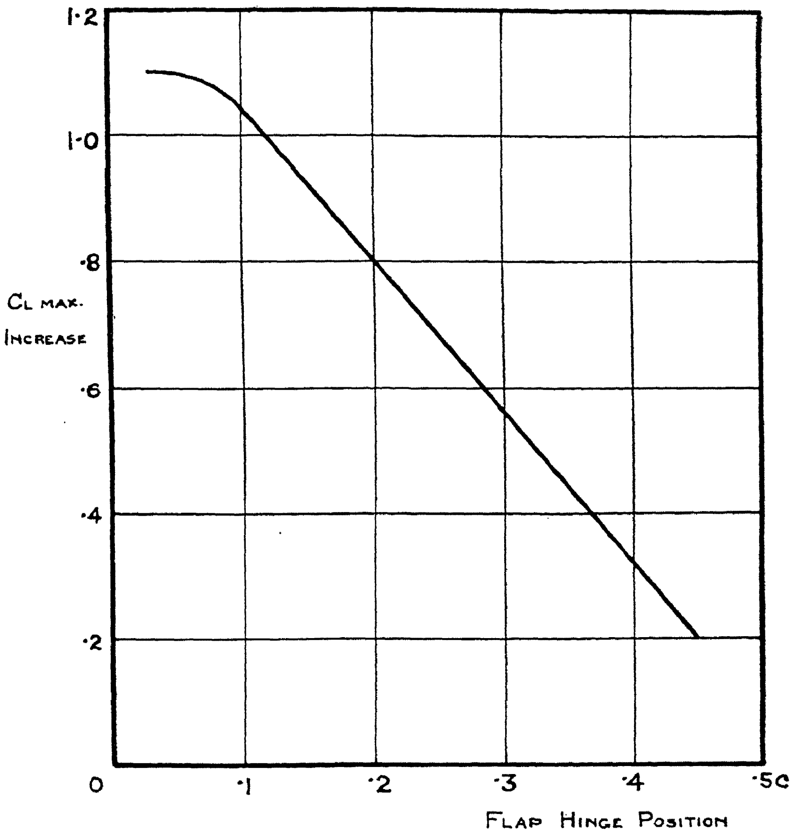
(0·2 CHORD FLAP)
FIG. 97.—EFFECT OF FLAP ANGLE VARIATION

The comparison of flaps situated over the central portion of a wing's span with outboard, or tip, flaps has shown the former to give greater increase in maximum lift coefficient, no doubt due to lesser end-losses, whilst the latter show a steeper gliding angle. Figures for flaps of chord 0·2 c, depressed at 60°, and running half the span, are as follows :

	C_L max increase	γ increase
Central flaps	0·54	5°
Tip flaps	0·34	7°

When considering the effect of partial span flaps it should be remembered that lift must be concentrated over the flapped

portion, with increase of downwash and lateral flow, so that the lift distribution over the unflapped portion does not remain the same as for a completely unflapped wing at similar incidence. Centre section flaps produce a span lift grading not altogether unlike that pertaining to a tapered wing,* with some increase



(FLAP CHORD 0·2c)
FIG. 98.—EFFECT OF FLAP HINGE POSITION

of the effective incidence towards the extreme wing-tips which may cause early tip-stalling.

The increase in $C_{L\ max}$ due to split flaps depends on the aerofoil section and is generally proportional to the $C_{L\ max}$ for the section. An undesirable feature of the split flap is the sudden collapse of lift at the stall and for this reason it is obviously of importance

* See Chap. VI.

that the stalled condition, with flaps down, should be carefully avoided by pilots.

Flight at a speed of, say, 15 to 20 per cent. greater than the stalling speed with flaps down, i.e., during the approach to land, takes place at a smaller angle of incidence than for a similar condition without flaps. This should prove beneficial as regards pilot's view.

Experiments* carried out with flapped, tapered wings have shown that for best results the flap chord should also taper from root to tip. This of course is only in agreement with the finding, already referred to, by which a flap chord of about 20 per cent. of the main wing chord was found to be most beneficial. Another feature of the tests is that the stall occurs at progressively lower angles of incidence with increasing flap deflection in the case of a tapered wing, a characteristic not present with rectangular wings.

The higher lift values of split flaps compared with the simple flap is no doubt due chiefly to the pronounced depression within the flap jaw, though it may be partially accounted for by the reduction of the horizontally projected area of the plain flap when in the down position.

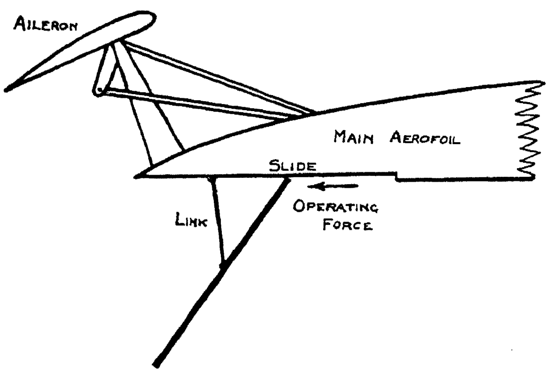


FIG. 99.—ZAP FLAP AND AILERON

The Zap Flap.

One of the more recent inventions for producing higher lift coefficients is the Zap balanced flap, of American origin.

* "The Effects of Full Span and Partial Span Split Flaps on the Aerodynamic Characteristics of a Tapered Wing," by Carl J. Wenzinger, *N.A.C.A.* Tech. Note 505.

This consists essentially of a flat surface cover to the rear underside of a wing which may be moved backward and downward (see Fig. 99), for the purpose of decreasing the landing speed and increasing the angle of glide.

At small flap angles there is increase in $\frac{C_L}{C_D}$ values, and some improvement of the take-off characteristics of aircraft fitted with Zap flaps is therefore possible. The arrangement reduces the operating loads, the hinge position of the supporting link, about which rotation of the flap takes place, being situated close to the flap C_P.

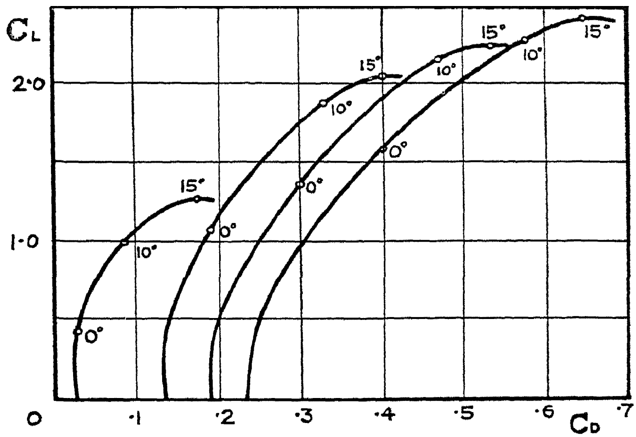


FIG. 100.—POLAR DIAGRAM FOR AEROFOIL WITH ZAP FLAPS DEPRESSED 0°, 30° 45° AND 60°

The aerodynamic characteristics of an aerofoil, Clark Y, with Zap flaps,* extending over the rear 30 per cent. of the main-plane chord, depressed at angles of 0°, 30°, 45° and 60°, are illustrated by polar diagrams in Fig. 100, from which it will be seen that $C_{L\,max}$ has increased from 1·3 to 2·4, and C_D from 0·1752 to 0·651 with the flap at an angle of 60°.

Unfortunately the added constructional complications of the Zap flap mechanism detract from its otherwise excellent qualities.

Fowler Flap.

This consists of what may be regarded as a small aerofoil housed within and below the main wing and which can be slid

* *N.A.C.A.* Technical Note 422.

backward and downward (Fig. 94). The increase in $C_{L\ max}$ reckoned on the normal wing area, and with a flap chord $0{\cdot}2\ c$, is 1·1; 1·5 for $0{\cdot}3\ c$; and as high as 1·9 for a chord of $0{\cdot}4\ c$, though when reduced to the extended area of main wing and flap these figures become 0·9, 1·1 and 1·3 respectively. These lift increments are high, even when reckoned on the extended area, but their usefulness is to some extent marred by the large backward movement of the C_P, which is of the order of $0{\cdot}2\ c$, and the serious pitching moment resulting. In one instance the flap is interconnected with the tail trimming device for safety, though the Fowler flap is not yet greatly employed.

Auxiliary Trailing-edge Aerofoil Flap.

This device, known as the Junker Flap, consists of an auxiliary aerofoil of symmetrical section which is mounted permanently near and below the trailing-edge, and which may be rotated about a hinge. Tests carried out in America* on a similar arrangement, known there as the Wragg compound wing, with a flap having a chord $0{\cdot}15\ c$ and hinged at a point on the auxiliary aerofoil chord 20 per cent. back from its leading-edge, showed the exact location of the hinge position relative to the main wing (Clark Y) to be of great importance, and substantial increased $C_{L\ max}$ possible only with the flap axis within a very limited region. The best position among those tried during the tests was found to be at $0{\cdot}0125\ c$ behind the main wing trailing-edge and $0{\cdot}025\ c$ below, in which position the flap touches the main trailing-edge when at an angle of 45° to the main-plane chord. In this position $C_{L\ max}$ was increased to 1·81 based on the combined wing area and to 2·08 reckoned on the original wing area, representing a 45 per cent. increase over that obtained with the plain wing. $C_{D\ min}$ was decreased from 0·0155 for the plain wing to 0·0146 with a flap deflection of $-5°$, thus showing favourable interference at low angles, but there was some diminution in rate of climb of roughly 10 per cent. The effect on lateral stability, determined by finding the angle at which autorotation commenced, was found to be slight.

Flaps and Ailerons.

In the case of simple trailing-edge flaps their situation may

* "Wind Tunnel Tests of a Wing with a Trailing-Edge Auxiliary Aerofoil used as a Flap," *N.A.C.A.* Technical Note No. 524.

be limited to that part of the span between the aileron roots, or the ailerons may be made to serve also as flaps by arranging for both simultaneous and differential action. Lateral control in this case is poor and it is doubtful whether the added complication is worth while.

Where split flaps run the entire length of the wing the conventional type of aileron is no longer possible. One alternative, known as the "Park bench," and illustrated in Fig. 99, consists of an additional surface mounted above the trailing-edge. The exact location of this type of aileron and its zero setting have been found to be very critical, but on the other hand the arrangement provides increased rolling moment coefficients with the flap down, a feature of value at slow speeds. The additional drag and weight, together with the undesirable torsion set up by the air loads on such an aileron, greatly detract from its usefulness.

Still another attempt to overcome the aileron problem associated with full span split flaps consists of normal ailerons situated over the flaps and which are restricted to upward movement only. Tests* have shown these upper surface ailerons to be quite satisfactory with the flaps in the neutral setting and to give positive yawing moments for large deflections, but with flaps extended, and above the stall, the rolling control is distinctly unsatisfactory, whilst the control forces necessary for operating such ailerons are too great for practical use.

Effect of Flaps on Longitudinal Trim and Control.

It would be thought that a down-turned flap must be accompanied by a considerable backward shift of the C_P, and thus produce an undesirable negative pitching moment. In fact the C_P shift is of the order of 0·1 c, but the effect of this on an aeroplane in flight is largely overcome by the increased downwash over the tail. Flight tests with large chord Zap flaps (0·28 c) have shown that the transition from a glide with flaps closed to one with flaps open is fairly smooth and may be made without use of the elevators.

Test results appear to show that the change from a glide at a speed of 1·2 V_s with flaps closed to flaps open requires an additional pitching moment of 0·04, and this could be obtained

* "Upper Surface Ailerons on Wings with Split Flaps," *N.A.C.A.* Tech. Report No. 499.

by changing the tail setting 2°, or by a down movement of the elevators of about 3°. If no tail alteration is made the flap-open glide would take place at the lower forward speed of about $1{\cdot}1\ V_s$.

The small change in trim necessary when flaps are brought into use, some tests at small incidences even having shown nose-up moments to be present, indicates that the downwash is greater than would be calculated from ordinary assumptions, especially since the air velocity over the tail is diminished by the presence of the open flaps. This pronounced downwash is confirmed by smoke tunnel tests made by the author and shown illustrated in Fig. 7.*

The change of pitching moment coefficient has been shown† to equal approximately one-quarter of the increase in the normal force coefficient, from which result it appears that the increment of lift due to flaps may be considered as concentrated at a point $0{\cdot}5\ c$ from the main plane leading-edge.

Flap Operating Forces.

Tests so far carried out indicate that the hinge moment for a split flap is roughly proportional to the angle of deflection. Roughly the flap moment is equal to $\frac{2}{3}\ x^2cW$ ft. lb., where $x = \dfrac{\text{flap chord}}{\text{main-plane chord}}$. Thus for an aeroplane weighing 600 lb., with a 4·75 ft. chord and $0{\cdot}2\ c$ full span flaps, the flap moment is 76 lb. feet ; a 3,000-lb. aircraft, with an 8-ft. chord and $0{\cdot}15\ c$ flaps, gives a moment of 360 ft. lb., whilst if the flap chord is increased to $0{\cdot}3\ c$ the moment becomes 1,440 ft. lb.

From the above it is seen that only very light aircraft, or aircraft with the flap chords no greater than $0{\cdot}1\ c$, can be equipped with manually operated flaps and that a small flap chord is very desirable for keeping the operational loads reasonably low.

By sealing the rear of the flap jaw and leading ducts to positive and negative regions of the wing's surface it becomes possible to operate the flaps pneumatically.

Balanced Split Flaps.

In order to reduce the loads required for operating split flaps the Miles flap has been designed in which one or more segments

* See p. 20.

† "Flaps and Pitching Moments," A. E. Russell, B.Sc., *Flight*, 29th Oct., 1936.

of the flap have the opening facing forward but are interconnected with the remaining flaps so that the air-stream tends to force open the former and thus assist in opening the others. The disadvantage of this arrangement is that the greatest hinge moment on the forward flap occurs very early, whereas it has been seen that with the normal flaps the hinge moment becomes progressively greater. Yet another device for giving partial balance is the Irving double hinge flap shown in Fig. 101, in which the flap is in two portions hinged at A and B, and is operated by the link CB being caused to slide horizontally at C.

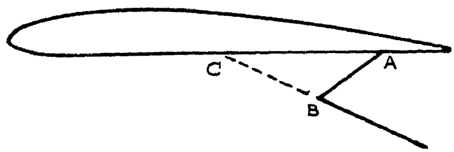


FIG. 101.—IRVING BALANCED FLAP

Flap gears have been devised in which their action is automatic, the flap position being made dependent on the C_P travel.

Summary of Flap Characteristics and Conclusions.

The conclusions arrived at in a report compiled at the R.A.E., Farnborough,* sum up the relative merits of the various flap arrangements as follows:

1. Split and slotted flaps give lift increases of the same order, but the increased drag is less with a slotted flap and hence it is better for take-off but worse for landing.
2. The utility of flaps for take-off depends on the thrust/weight ratio from rest to take-off speed. If this value is between 0·2 and 0·3 only a slotted flap is of use, but over 0·3 almost any flap gives some saving. Greater distance can be saved by the aid of flaps with low thrust values than with high values.
3. Increased load at take-off of 5 lb. per sq. ft. with full scale slotted or Fowler flaps if the ratio $\frac{T}{W} = 0{\cdot}25$. If the value is 0·4 the increase in load amounts to 20 lb. per sq. ft. (The take-off distance is taken as 1,500 ft.) These figures need confirmation by flight tests.

* "Trailing-Edge Flaps in Relation to Take-off and Landing of Land-planes," *R. & M.*, No. 1707.

4. Up to a value of 20 lb. per sq. ft. either a split or slotted flap is suitable for landing, but up to 30 lb. per sq. ft. a large size Fowler flap is recommended for lower landing speed but with flatter glide.
5. The main advantage of the Fowler flap is the high $C_{L\ max}$ and consequent lowering of V_{min}. It is useful for getting loads of over 20 lb. per sq. ft. off and on to the ground.

The drag order of the various devices being, in order of efficiency, slotted flap, plain flap, Zap, and Schrenck indicates the advantage of the slotted flap for take-off and of the Schrenck for landing purposes. An investigation carried out in America* of the effect of flaps on take-off showed that substantial reductions in the take-off run are possible, provided the proper flap angle for the conditions obtaining is employed. The optimum angle varies inversely as the power loading and to a smaller extent with wing loading. The Fowler and the auxiliary aerofoil flap appear to give the shortest take-off run, for which purpose a lift coefficient value of about 78 per cent. of the maximum should be used, regardless of the wing loading. With all other devices the lift coefficient for shortest take-off lies between 82 and 89 per cent. maximum.

Increased lift coefficients by the aid of leading-edge slots, with or without trailing-edge flaps, are obtained only by increase of incidence which is undesirable. If the slot opening is automatic it does not come into operation until too low a speed has been reached and may upset the process of flattening out.

A possible flap combination of promise is one in which the take-off is accomplished with a slotted flap, whilst for landing either a plain, unslotted flap, or a split flap is used. The provision of a slot during take-off only should not be difficult to arrange for.

The employment of flaps on biplanes has not been found so beneficial as with monoplanes, due no doubt to the interference effect of the upper wing flaps. Best results have been obtained with the top wing flap arranged to open to half the angle of that for the bottom wing, though it is doubtful whether the small gain in lift from the upper wing of a biplane is worth the extra weight and complication.

In conclusion it has been seen that for wings having full

* *N.A.C.A.* Tech. Note No. 568.

span flaps, or at least where the greater part of the span is flapped, the designer may choose a section of small mean camber, possibly of 2 or 3 per cent., and thus gain the advantage of low C_D values for high speed, whilst relying on the flaps to give the desired high lift values at slow speed. Thus we shall probably see aerofoil sections developed to give very low drag at small values of lift, this being the criterion of paramount importance, whilst $C_{L\ max}$ will become a property associated almost solely with flaps.

Thurston Wing-tip Rotor.*

The wing-tip rotor is similar to the slotted wing, but the auxiliary aerofoil is mounted at its centre on a vertical spindle set perpendicularly to the chord line (see Fig. 102). In action the rotor moves forward and upward from the main wing and revolves by autorotation, thus deflecting the air-flow down on to the upper surface of the main aerofoil.

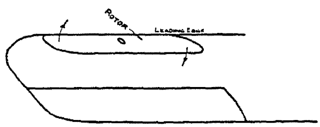


FIG. 102.—THURSTON WING-TIP ROTOR

It is claimed that by this means the C_L values are increased by 48·5, 48, 71 and 75 per cent. at angles of 10°, 15°, 20° and 25° respectively. The chief virtues of this device are that higher lift coefficients can be obtained at the smaller angles of incidence, which is not possible with ordinary wing slots, and the lift coefficient curve retains a fairly constant value over the larger angles, say from 15° to 25° incidence, so that although the speed of flight may be too low to enable the full lift to be obtained, at least unstalled conditions will be retained and lateral control be still available.

* *R.A.S. Journal*, Jan., 1934.

CHAPTER XI

MANŒUVRES AND DYNAMIC LOADS

IT is not intended here to give a full explanation of aircraft pilotage, but instead to state briefly how the most common manœuvres are carried out and to note their effects on the loading to which the main air-frame structure is subjected.

In steady horizontal flight the load on the wings is equal to the total weight of the aircraft, which means that the main plane incidence is adjusted for each speed of flight to give that value of C_L for which $C_L \frac{\rho}{2} SV^2$ is equal to the total weight. That is to say, as V decreases C_L must increase to balance, until at the slowest flight speed, $W = C_{L\,max} \frac{\rho}{2} SV^2_{min}$.

Manœuvres, however, can only be accomplished by the introduction of accelerated motion, during which the lift does not remain equal to the weight, but may become several times as great, or even less.

Suppose, for example, the angle of attack of the main plane is suddenly increased during horizontal flight to give twice the original C_L value with inappreciable loss of speed. Then the lift must be increased, temporarily at least, to twice the weight, which means that the aeroplane commences to climb.

The Take-off.

It is advisable, and often essential, that the take-off should be accomplished with as short a run as possible, and for this reason it should always be made "into wind," i.e., in the direction opposite to that of the wind, unless other considerations, such as shape of aerodrome, general slope of surface, or the presence of obstacles, render preferable a down-wind, or side-wind take-off. This applies likewise to the landing of aircraft.

Starting from the leeward side of the aerodrome, and facing directly into wind, the engine throttle is opened fairly quickly

to the full-open position, and the machine commences to move over the ground. The control stick is placed in the neutral position so that no extra drag is caused by the elevators. Quite shortly after the start, when it is judged that the speed is sufficient for the tail to become self-lifting, the stick is eased forward so that the tail-skid, or wheel, comes clear of the ground, with the fuselage practically horizontal.

In this position the resistance to forward motion is least, and acceleration is consequently greatest. When a speed somewhat in excess of stalling has been reached* the control column is eased gently back so that the wheels come clear of the ground, when the stick is immediately returned to, or near, the neutral position. Extra speed is then rapidly, and safely, gained prior to commencing the initial climb.

If the field, or length available for the run, is short, the climb is commenced as soon as a forward speed slightly above stalling is attained and a gentle climb (keeping careful watch on flying speed) is maintaine until all obstacles are cleared, when the nose may be slightly depressed to obtain a greater margin of speed prior to re-climbing.

The dangers of a climbing turn just after taking-off will be realised, since it has been seen that a turn, without loss of height, can only be accomplished by means of increased lift, which is not generally available at this stage. All other manœuvres should likewise be avoided when taking off, since any displacement from neutral of the control surfaces must inevitably cause extra drag, and so tend to retard speed.

The structural loads during a properly executed take-off are comparatively small. The wing loading gradually increases from zero to unity and increases further according to the rate of upward acceleration.

The Dive.

The forces acting on an aeroplane in a dive are fairly similar to those for normal flight. As the speed is increased so the value of C_L decreases, and this is accompanied by a rearward movement of the C_P with a compensating down-load induced on the tail-plane.

If the dive is allowed to become steeper, the speed continues to increase until the drag is equal to the total weight, when

* See p. 173.

conditions of equilibrium are reached. This is called the *limiting nose dive* (L.N.D.), or the *terminal velocity dive* (T.V.D.), since no greater velocity is possible.

The flight path then is roughly vertical with a wing incidence just above the no-lift angle, and a " down " load on the tail to counteract the main plane reaction moment.

The L.N.D. velocity varies from about 200 m.p.h. or less for quite light aircraft to perhaps 750 m.p.h. for well streamlined machines with high wing loading. Average values range from about 300 to 400 m.p.h.

As an aeroplane is pulled out of a steep dive, the wing incidence and hence C_L value are increased. Owing to the very high speed of flight, it is easily seen that high lift values may be imposed on the wing structure, and such change of attitude, if applied suddenly, may be sufficient to cause collapse of the structure unless the aircraft has been specifically designed to withstand such a manœuvre. For example, an aeroplane having a minimum speed of 60 m.p.h. if pulled out suddenly from a dive at 300 m.p.h. would be subjected to a loading of $\frac{(300)^2}{(60)^2}$ W $=$ 25 W. This is based on the assumption that there is momentarily a negligible loss of speed and that during the manœuvre the wing incidence is held at the $C_{L\,max}$ angle. The result of tests in actual flight justify these assumptions in general.

Since it is impracticable generally to build aircraft to withstand such heavy loads, care should be exercised during the recovery from steep dives. One method of ensuring this is to so design the elevators that rapid recovery is impossible. In this connection it will be remembered that unbalanced control surfaces are not so readily actuated as balanced surfaces.

The Zoom.

Zooming is accomplished by changing from a dive to a steep climb. If the zoom is prolonged the machine is spoken of as " hanging on its propeller " and the effect is, in fact, an approach to helicopter flight. The loading on an aircraft can be very high during this manœuvre, and depends on the rapidity with which the change-over from diving to climbing flight is accomplished. The case is similar to the previous case, but the top speed is not likely to approach the terminal velocity.

The Loop.

A loop consists, as is well known, of a complete circular rotation in a vertical plane, so that the path of flight describes a loop. With moderately slow aircraft, speed is increased prior to the loop by an initial dive. The climb is made at a large angle of incidence, both speed and incidence decreasing towards the top of the loop, where the machine assumes an inverted position.

After passing the top of the loop, the aircraft is dived, and thus regains speed before being pulled out into level flight again.

The loads set up during loops vary considerably according to the manner in which the manœuvre is executed, but in all accelerometer records there are two pronounced "humps," as shown in Fig. 103, which indicate the incidence of loading during the initial climb and again during the pull-out from the resulting dive. The points 1–4 on the diagram denote the initial dive; the climb; inverted position; and the pull-out from the dive.

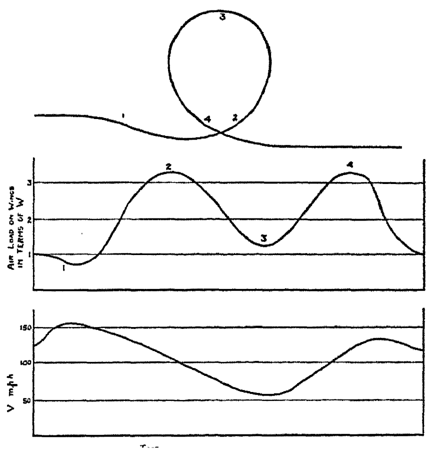


FIG. 103.—THE LOOP

In well-executed loops the lift acceleration reaches a figure of from 2 to 4 g, but may become as high as 6 to 7 g with harsh use of the elevator. The loading is outward throughout, that is to say the mass forces act towards the floor of the fuselage, and the loading on the wings is therefore normal; the incidence being positive.

If the loop is carried out slowly, it is possible to "hang" in the inverted position. In this case the outward, or centrifugal, loading decreases to zero, and the loading on the main planes reverses, so that they support the aircraft in inverted flight. Even during a normal loop it is not unusual for the loading at the top to fall below unity (pt. 3 of Fig. 103).

Inverted Flight.

The flight of an aircraft in an inverted position does not really

differ aerodynamically from normal flight, except that the aerofoil is generally inefficiently shaped for such a condition. The setting of the wing to the fuselage also is inefficient in inverted flight, and results in a large angle of the fuselage axis with the horizontal. This angle is made up of the sum of the fixed wing incidence, and the inverted aerofoil angle of attack necessary to produce the required lift at the flight speed obtaining.

The high drag of an inverted aerofoil, together with the high drag of the fuselage in its tilted position, results in a slow forward speed, for which the C_P is at, or near, the C.P.F. position. Obviously it is impossible to attain high wing loadings in the inverted position if high speed is impossible of achievement, and it is because of this that a loading factor of $\frac{2}{3}$ normal C.P.F. is called for in strength calculations.

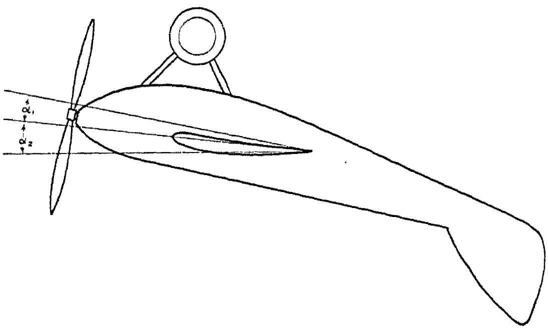


FIG. 104.—INVERTED FLIGHT

The inverted loop, or bunt, may be regarded as inverted flight inasmuch as the loading of the main planes during part of the loop is negative. Large loads may be thrown on the structure during this manœuvre, but unless the aircraft has been specially designed for such flight, it is not considered a normal evolution.

During inverted flight the anti-lift bracing system comes into operation in place of the lift bracing, whilst of course the spar loadings are also inverted, so that the compression flange becomes the tension flange and *vice versa*.

The Stall.

There is no generally agreed definition of stalling, but it may be said to be the condition that exists at the minimum forward speed, when the aeroplane commences to "pancake," i.e., when

the lift no longer equals the weight, and horizontal flight becomes impossible.

Stalling takes place when the wing incidence reaches and just passes the maximum C_L value, and is usually coincident with the setting in of burbling of the air-flow.

The aircraft may, or may not, be under control. Control may be maintained by arranging for the ailerons to remain effective after the commencement of downward acceleration by (*a*) wash-out of the wing incidence at the tips, (*b*) the use of floating ailerons, or (*c*) slotting part of the wing forward of the ailerons. In each of the methods enumerated the lift may be less than the weight, so that the aircraft velocity has components both forward and downward, but by arranging for the ailerons to be unstalled, i.e., to be set at an incidence below that of $C_{L\ max}$, lateral control may be maintained.

If stalling is described as flight at an incidence coarser than that for $C_{L\ max}$ it becomes possible to stall a machine in high speed flight, but without loss of height. This is not, however, generally regarded as a stalled condition, though the wing is said to be at stalled incidence.

When effective lateral control is not present at slow speed or is not exercised, stalled flight develops into a spinning nose dive.

Turning.

This has already been dealt with in Chapter VII,* from which it was seen that, theoretically, the lift on the wings is given by $L = \dfrac{W}{\cos \phi}$, where ϕ is the angle of bank. Thus for a bank angle of 75°, the wing loading is approximately 4W.

It should, however, be noted that the fuselage, fin and rudder will contribute towards the supporting force, and that this component of lift increases as the angle of bank approaches the vertical. Furthermore, since the total lift of the wings cannot exceed $C_{L\ max} \dfrac{\rho}{2} SV^2$, there must be some limiting angle of bank for each velocity, beyond which horizontal flight is impossible. For instance, in a turn at a flight speed of three times stalling the maximum wing loading is $W\left(\dfrac{3}{1}\right)^2 = 9W$.

* See p. 88.

In practice, accelerations approaching 9 g have been measured.

Spinning.

This manœuvre may be carried out voluntarily, or by accident, and is brought about, initially, by stalling. When an aerofoil is at an incidence just below the stalled attitude, a slight span-wise variation in the angle of attack, due to aileron movement, or disturbed air-flow, causes a considerable difference in the forces acting on the two wings. For, consider an aeroplane with its wings at an incidence relating to the C_L value shown at A, Fig. 105, and suppose actuation of the ailerons so to change the incidences of the two wings that the mean C_L values of each wing are as given by points B and C, where C is at an angle greater than that for $C_{L\,max}$. The result is twofold, the average C_L value for the complete wing is decreased, with a corresponding loss of lift, whilst the drag of the two wings is proportional to the C_D values as represented by points E and D. The lift values of the left and right wings will probably vary little, but drag, on the other hand, will be much larger on the wing with the coarser incidence (down-going aileron).

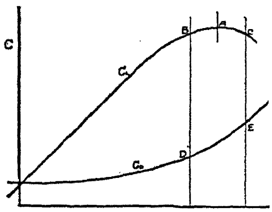


FIG. 105.—AUTOROTATION

Hence the aeroplane starts to lose height and at the same time commences to rotate, or spin.

Immediately the spin starts the outer wing travels at a faster speed than the inner wing, whilst, of course, its incidence is decreased owing to the upward movement (relative to the flight path) of that wing, so that the tendency is for it to remain un-stalled. Similarly, the inner wing becomes more stalled. For the wings alone a steady condition is thus set up, known as auto-rotation.

The tail-plane and rudder act against the spin, and tend therefore to counteract the spinning, according to the principles of ordinary stability. If it is desired to remain in a spin it is usual to raise the elevator, to maintain the stalled incidence of the main plane, and to move the rudder over to one side (the trailing side) so as to overcome its tendency antagonistic to the spin. The initial spin then develops into a flat spin in which the whole wing becomes completely stalled at an angle of incidence of roughly 40°, and recovery is then difficult.

The flight path takes the form of a helix with a vertical axis. Owing to the stalled condition, the speed along the spiral path is low and hence the speed of descent is relatively low. The duration of each complete revolution varies according to the design of the aircraft, being generally about two to three seconds, but may be increased by three or four times with slotted wings, whilst height is lost at the rate of 1,000 ft. in from four to eight turns. Thus an aeroplane spinning at the rate of two-second revolutions, with a height loss of 1,000 ft. for every six complete turns, has a vertical rate of descent of 57 m.p.h.

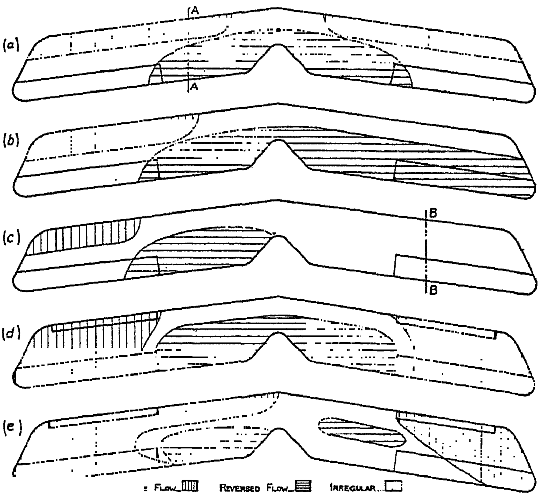


FIG. 106.—AIR-FLOW OVER WING DURING A SPIN

Fig. 106 illustrates the condition of air-flow over the upper wing (top-side) of an "Atlas" in spinning flight.*

In the first diagram, (*a*), the air-flow is as illustrated in Fig. 5 (*e*), in which the air passing under the wing turns upward beyond the trailing-edge and then flows forward over the upper surface for some distance. Stalled conditions appear to be setting in from the centre of the wing and developing outwards.

In the second diagram, (*b*), stalling has been followed by a right-hand spin. It will be noticed that the outer (left) wing conditions are not greatly altered, but that the inner wing is stalled from root to tip. Two seconds after the spin commenced,

* "Airflow about Aeroplanes shown by Wool-tufts," *R. & M.*, No. 1494.

(c), the air-flow over the inner wing is seen to fluctuate between pattern (a) and pattern (b) of Fig. 107.

Diagrams (d) and (e), Fig. 106, show the air-flow over the wing with slots open, from which it is noticed that both before and during the spin, the air-flow is smooth over the slotted portions, but that over the centre, unslotted length turbulent conditions are present, particularly during the spin.

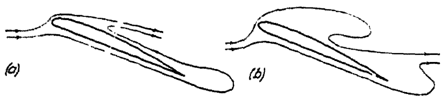


FIG. 107.

Recovery from a spin is effected by centralising the ailerons and setting the rudder against the spin to stop rotation. Depression of the elevators is often helpful in decreasing the wing incidence and so re-establishing unstalled conditions. Flying speed is gained in the resulting dive, after which the aeroplane may be brought to the horizontal flight position. In some instances "opposite" aileron control is used for helping to stop the spinning motion, but this is generally of little avail and may result in a reversal of the spin direction.

The effectiveness of the tail surfaces is reduced during a spin, due to the air-flow having been disturbed by the passage of the stalled wings, whilst also the rudder and tail-plane tend to blanket each other to some extent.

The acceleration during a spin varies between 2 g and 3 g. (See also Appendix.)

The Roll.

A roll consists of a complete rotation about the longitudinal axis with that axis remaining roughly horizontal. Concurrently with the commencement of rolling by means of the ailerons, the elevators are raised, which causes an initial climb and thus compensates for loss of height. A second climb is made during the last quarter of the revolution to bring the machine back on to the original path, although some reduction in speed is occasioned by the manœuvre.

The loading during the initial climb is high, being in the nature of from 5–7 g, according to the rapidity of roll and commencing speed, whilst the second climb causes an acceleration of from 2–4 g. A high acceleration value during the roll is not surprising since the manœuvre is a combination of both climbing and turning.

The Side-slip.

As the name suggests, this consists of a sideways slip from the forward flight path, and is effected by banking the aeroplane, with simultaneous application of "top" rudder to prevent the nose from falling due to directional stability.

The velocity consists of the forward and sideways components, and since the drag value in this position is high it is possible to lose height rapidly without gain of forward speed. The steepness of slip may be raised at will.

The side-slip thus provides one of the simplest and safest methods of losing height prior to landing, especially where the landing ground is bounded with high buildings, or trees, and may be continued until within a few feet of the ground, followed by a final turn into wind just before making contact. It may be regarded as one of the most useful of all manœuvres.

During a side-wind landing, when the wind carries the aircraft sideways across the ground, a gentle side-slip into wind will counteract the sideways movement, and enable contact to be made free from lateral drift and consequent wrenching at the wheels.

The pilot's view during side-slipping is generally good. Alternate left and right slips may be made in order to prevent the aircraft from moving too far to one side of the landing field.

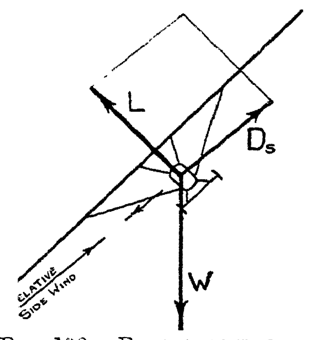


FIG. 108.—FORCES ACTING IN A SIDE-SLIP

The loads set up on an aircraft during a side-slip are not likely to be high, since the weight is supported partly by the main plane lift and in part by the drag due to the sideways motion (see Fig. 108). The side loads on fin, rudder and fuselage, however, may be appreciably high.

Landing.

Landing is normally commenced by a glide with engine throttled; the speed of flight being roughly 20 per cent. above stalling, or minimum speed. Side-slipping, just described, may be resorted to for accelerating the descent.

Within a few feet of the ground, the speed is gradually decreased, by pulling back the control stick, so that the loss of forward speed is accompanied by increased incidence. If the approach is correctly judged, the main plane incidence corresponds to the maximum lift coefficient and minimum speed just as the wheels and tail skid make contact.

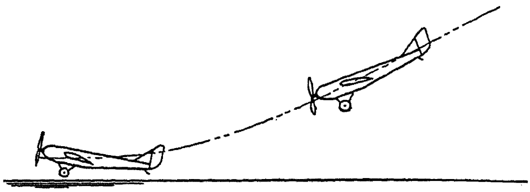


FIG. 109.—LANDING

The loading induced during such a "three-point" landing does not exceed 2 g, but if the levelling off is left until too late, so that contact is made with the wheels alone, the loads imposed may be as high as 5 g, or even higher according to the speed of descent. On the other hand if stalling speed is reached when still some feet above the ground, the machine is said to "pancake," and again the loads may be high.

An analysis of 340 manœuvres carried out with 14 different types of aircraft at Martlesham Heath* provided the acceleration figures given in Table III. The conclusion arrived at from this analysis is that for correctly performed manœuvres an average maximum acceleration of 3 g may be assumed, but that considerably higher values are possible if the aircraft is improperly handled.

Other tests recorded in America† and elsewhere show higher values than those of Table III, a figure of 5·5 having been measured in a power spiral, 4·4 in an Immelmann turn, 5·7 in a vertical bank at 150 m.p.h. only, with 6·1 in a rocket loop at

* "Measurements of Accelerations on Aircraft during Manœuvres," *R. & M.*, No. 1392.

† *N.A.C.A.* Report No. 203 (1924).

160 m.p.h., whilst a sudden pull-out from a dive has produced an acceleration as high as 10·5 g at a speed of 173 m.p.h.

TABLE III.—MAXIMUM ACCELERATION DURING MANŒUVRES

Manœuvre	Max. Acceleration in terms of "g"	Notes
Rocket loop	2·95	Entry of loop
Normal loop	2·8	,,
Half loop and roll out	2·9	
Flick roll	3·1	
Slow roll	2·5	
Upward roll	2·9	
Half roll and dive out	2·8	
Dive and pull-out	3·15	
Turn	2·8	

Gusts.

The presence of disturbed air conditions results in dynamic loading in excess of normal. Measurements made in this country have given values as high as 2·5 g, whilst in America under very severe conditions an acceleration of 4 g has been measured.

Chapter XII

THE AIRSCREW

Engine Power and Torque.

Before considering the airscrew, it is as well first to make a brief examination of the nature of the engine power available. Fig. 110 gives typical curves, as supplied by the engine manufacturers, for b.h.p. plotted against r.p.m., from which it is seen that the horse power increases more or less in direct proportion to the engine speed.

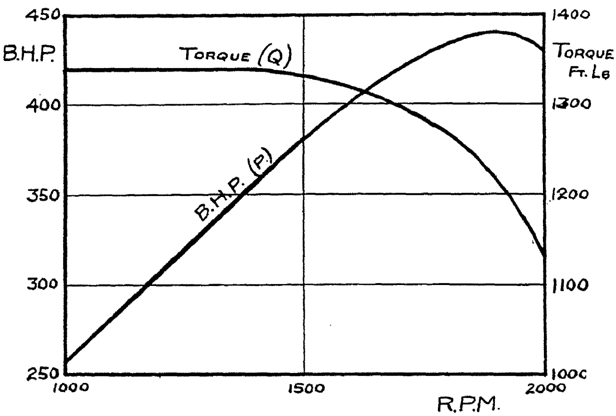


Fig. 110.—Engine Power and Torque Curves

The turning effort, available for rotating the airscrew, is shown by the torque curve, and is obtained from the formula

$$Q = \frac{550\ P}{2\ \pi\ n} \text{ in ft. lb} \qquad (20) \qquad \begin{array}{l} \text{where } P = \text{b.h.p.} \\ \text{and } n = \text{r.p.s.} \end{array}$$

It will be noticed that the torque, or turning effort, of the engine falls off slightly with increase of engine speed, despite the fact that the power is increasing. This means that the work done per revolution, $2\ \pi\ Q$, is slightly decreased at higher engine

speeds, due to greater frictional losses, but that the work done per unit of time, $2\pi n Q$, the measure of power, increases. Within the engine speed range used in practice, the torque may be assumed roughly to be constant.

The purpose of the airscrew is to convert the torque of the engine into forward thrust. Each blade is an aerofoil, which moves along a helical path, with rotational and translational velocity components. The rotational velocity of any point is equal to the circumference of the circle through which it turns multiplied by the number of revolutions per unit of time, say πDn ft. per sec. at the tip, where D = airscrew diam. in ft. See Fig. 111.

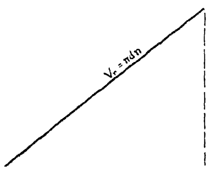


FIG. 111
ROTATIONAL VELOCITY OF AIRSCREW

The translational velocity, V_t, for all points on the airscrew, is equal to the forward speed of the airscrew relative to the air, or roughly equal to the aircraft speed in normal flight.

Blade Element or Strip Theory.

It has been seen that the speed due to rotation is not constant, but varies along the length, or span, of the blade, since the path of rotation traversed changes from zero at the hub centre to a maximum of π D ft. per rev. at the tip. For this reason, the blade may be considered as made up of a number of small strips, each of which can then be treated as a separate aerofoil having its own particular speed. (Strips are assumed so narrow that variation of rotational velocity over the width is negligible.)

The total force, R, acting on the element of blade is divided, for convenience, into two components, R_T and R_Q, *parallel and perpendicular to the direction of movement of the aeroplane.* (Note that R is not resolved parallel and perpendicular to the direction of the air-flow relative to the blade, as in the case of aerofoil drag and lift.)

The parallel, or axial component is called *thrust,* and the perpendicular, or radial component is called *torque reaction.* A certain similarity to lift and drag of an aerofoil will be recognised. The thrust component is the useful part of the total force, whilst the torque reaction is the price paid for conversion of power, in the same way that drag of a wing is the price paid

for lift. The summation of all components R_T gives the thrust, T; whilst the components R_Q, multiplied by their respective distances from the hub centre, give the airscrew torque, Q.

The airscrew torque, it will be seen, opposes the turning motion of the engine, and in fact tends to revolve the aeroplane in the direction opposite to that of the engine crankshaft. In other words, the tail wagging the dog. At any steady engine speed, the airscrew torque is equal to the engine torque.

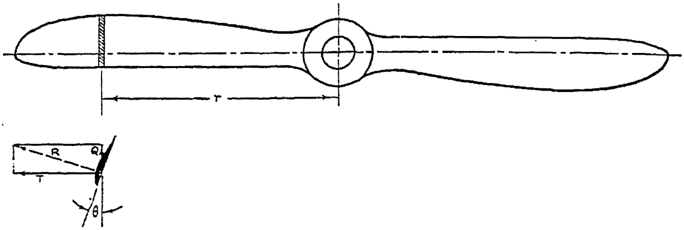


FIG. 112.—FORCES ON ROTATING AIRSCREW
(*Note.*—For T and Q read R_T and R_Q respectively.)

Curves of Thrust and Torque.

For simplicity, consider first the airscrew revolving with no translational, or forward, velocity, the angle of incidence in this instance being the blade angle. The total reaction on an aerofoil moving through the air has been seen to be $R = C_R \frac{\rho}{2} S V^2$. S represents the area of the wing surface for weight-lifting aerofoils, but for the purpose of airscrew calculations it is more convenient to use the "swept" area, $\frac{\pi}{4} D^2$ or simply D^2 and to modify the coefficient, C_R, accordingly. Also $V_r = \pi n d$, and varies from zero to $\pi n D$.

Hence the resistance equation may be written

$$R = C_R \frac{\rho}{2} D^2 (Dn)^2 = C_R \frac{\rho}{2} n^2 D^4 \quad . \quad . \quad . \quad (21)$$

$$\text{Therefore } T = C_T \frac{\rho}{2} n^2 D^4, \quad . \quad . \quad . \quad . \quad (22)$$

and since the moment of a force is proportional to the lever arm,

$$Q = C_Q \frac{\rho}{2} n^2 D^5, \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (23)$$

FIG. 113.—DE HAVILLAND "ALBATROSS" AIR LINER
FOUR "GIPSY TWELVE" ENGINES

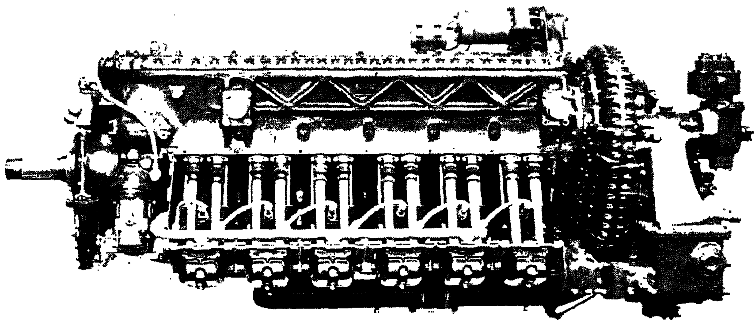


FIG. 114.—DE HAVILLAND "GIPSY TWELVE" ENGINE

[Facing page 156

where C_T and C_Q are thrust and torque coefficients for the airscrew section considered.

Translational Velocity Component.

If the airscrew is now made to move forward, as well as revolve, then a translational velocity, V_t, is superimposed on the rotational velocity, V_r, which alters the incidence of each blade element from θ to $\theta - \phi$ (Fig. 115). Values of R, and the coefficients C_T and C_Q will now depend on the resultant of the two component velocities.

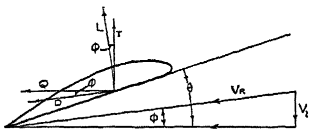


FIG. 115.—EFFECT OF COMBINED TRANSLATIONAL AND ROTATIONAL SPEEDS OF AIRSCREW

The resultant velocity, V_R, is seen to depend on both V_r and V_t, or on the angle ϕ, since $\tan\phi = \frac{V_t}{V_r}$.

Further, V_r is proportional to $n\ d$, and therefore the angle ϕ is proportional to $\frac{V_t}{n\ d}$, so that R is dependent on $\frac{V_t}{n d}$, and likewise are T and Q.

Notice that thrust and torque reaction are now given* by

$$R_T = L\cos\phi - D\sin\phi, \text{ and}$$
$$R_Q = L\sin\phi + D\cos$$

In the same way that aerofoil coefficients, C_L and C_D, are plotted against α, the coefficients C_T and C_Q for each airscrew section may be plotted against values of $\frac{V_t}{n\ d}$.

The ratio $\frac{V_t}{n\ d}$ is generally written as $\frac{V}{n\ D}$, where V is the forward speed of the aircraft. Fig. 116 shows the coefficients plotted against $\frac{V}{n\ D}$ for one airscrew. For example if V = 130 ft. per

* The air-flow over an airscrew blade is assumed to be two-dimensional, and hence values of C_L and C_D for the airscrew section are corrected for infinite aspect ratio.

sec., $n = 20$ r.p.s., and airscrew diameter $= 10$ ft. then the of $\frac{V}{nD} = \frac{130}{20 \times 10} = 0{\cdot}65$.

The maximum efficiency of a lifting aerofoil is obtained at the angle of attack giving $\frac{L}{D}$ max. Similarly with an airscrew the optimum results are obtainable when the angle of attack of the blades $(\theta - \phi)$ at all sections is set for $\frac{C_T}{C_Q}$ max

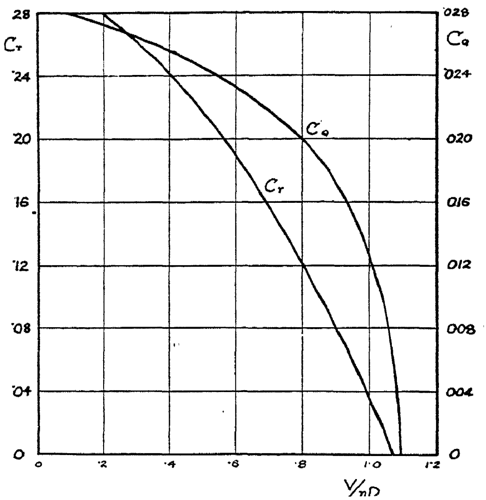


Fig. 116.—Airscrew Thrust and Torque Coefficients

at the flight speed considered most important. Generally this is top speed, or rather less in order to give an improved performance at the lower speeds of flight, but where top speed is chosen as the criterion for the blade angle setting, the loss at lower speed is not generally great.

It may be noted in passing that the angle for $\frac{L}{D}$ max is generally smaller for the high speeds of propeller work than for the slower speeds to which lifting aerofoils are subjected.

The statement that the angle of attack adopted is that which produces the highest value of $\frac{C_T}{C_Q}$ is not quite correct and needs some qualification. In making use of the thrust value

appertaining to $\frac{C_T}{C_Q}$ max, a smaller diameter would generally result, with higher slipstream speed and consequent high drag of parts within the slipstream cylinder. In view of this it is more usual to adopt a rather smaller angle of attack, giving on the average a C_L value of 0·6.

Relation between Speeds of Aircraft and Engine.

Consider an aeroplane in steady horizontal flight, in which the propeller torque and engine torque are equal, and then let the throttle be opened to give an increased engine speed. Suppose the r.p.m. for the engine of Fig. 110 are increased from 1500 to 1600, then it is seen that the torque changes from 1330 to 1316 ft. lb. An airscrew of 10 ft. diameter would give C_Q values of 0·018 and 0·0158 respectively for the two speeds $\left(\text{from } C_Q = \dfrac{Q}{\frac{\rho}{2}\, n^2 D^5}\right)$ and from the characteristic curves of Fig. 116 the aircraft speeds for these two conditions are seen to be 220 and 245 ft. per sec. respectively. In other words, by increasing the engine speed from 1500 to 1600 r.p.m. the aircraft speed is increased from 220 to 245 ft. per sec.

Conversely, as the aircraft speed decreases, the rate of rotation must fall so that the airscrew resistance torque remains equal to the engine torque, which latter has been seen to be roughly constant.

Thrust Variation.

The airscrew thrust has been seen to depend on the effective angle of attack of the blade element, and on the speed of revolution. For the maximum engine rotational speed, the effective angle of attack is decreased as forward speed is increased, and hence there must be some falling-off of thrust. In other words an airscrew with a fixed blade angle has its maximum thrust at one particular forward speed, beyond which thrust decreases, as shown by the airscrew thrust curve of Fig. 117. The speed at which the maximum thrust is made available may be chosen to suit the requirements of a good take-off, or more generally of a high top speed, but since for a fixed pitch airscrew both

conditions cannot be catered for in one design, the blade angle must be a compromise between the two requirements. For example, the airscrew may be stalled at zero forward speed in order to give a greater maximum flight speed, but only by rendering inferior the take-off conditions.

The curve of Fig. 117 represents the thrust available for all flight speeds from zero to the limiting velocity dive, where it has become negative, i.e., despite the rotation due to the engine power, the propeller causes a drag acting against forward motion. (Note that this does not necessarily mean that the effective

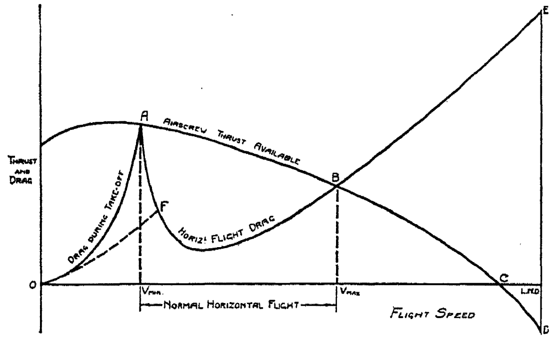


FIG. 117.—AIRSCREW THRUST AND AIRCRAFT DRAG THROUGHOUT FLIGHT RANGE

angle of attack of the blade is less than the no-lift angle. When the rearward component of Drag, D sin ϕ, becomes greater than the forward component of Lift, L cos ϕ, the thrust becomes negative.) Between A and B is shown the curve for drag in horizontal flight, between the minimum and maximum speeds. The excess of available thrust over drag may be used for climbing, but otherwise level flight can be maintained only by throttling the engine to some extent.

Beyond B speed can be increased only by depressing the nose of the aircraft in which case the acceleration due to gravity is assisting the waning airscrew thrust. The total aircraft drag increases rapidly with speed until the limiting velocity is attained, when the drag, represented by D E, is equal to the total aircraft weight, and no further increase of speed is possible. The curve O A gives the drag of the machine during the run prior to the take-off, if the machine is to be air-borne at the lowest speed of flight. More usually the take-off is made at a higher speed, i.e., the main plane angle of attack is kept below the maxi-

mum. The take-off is then at some point, F, where extra thrust is immediately available for a climb, if required.

Pitch.

The pitch of a screw is generally determined by the distance through which it would move when rotated in a solid medium, through one complete revolution. A better definition would perhaps be the relative axial movement of the screw and working medium for one revolution of the screw. In the case of an airscrew, however, owing to the lack of solidity of the working medium, the movement relative to the air varies under different conditions according to the amount of slip, so that the usual definition of pitch is not truly relevant here.

There are two pitch definitions used for airscrews ; the geometric and the experimental.

GEOMETRIC MEAN PITCH.—This is defined as the pitch of the helix forming the blade chord line, or, in other words, the distance parallel to the axis that an airscrew would move for one revolution, if the blades are imagined to rotate in a helical slot so that the underface of the blades slides along the face of the slot, or groove, similarly to a normal screw.

The blade, or geometric pitch, angle varies along the blade length, being a minimum at the tip and increasing towards the hub.

Thus geometric pitch, $p_g = 2\ \pi\ r\ \tan\ \theta_r = 2\ \pi\ R\ \tan\ \theta_R$, where θ_R and θ_r are the blade angles at the tip radius R, and at any other radius r. (Fig. 118.)

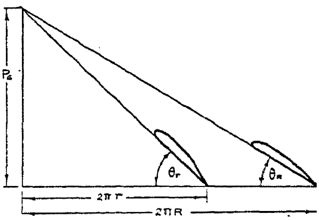


FIG. 118.—GEOMETRIC PITCH OF AIRSCREW

Geometric pitch, as measured at a point $\frac{2}{3}$ R from the hub centre, is used for reference purposes and the value is stamped on all airscrews. Because in some cases the pitch varies slightly along the length of the blade, the position $\frac{2}{3}$ R has been fixed as reference datum.

Hence the mean geometric pitch as defined, $p_g = 2\ \pi \cdot \frac{2\ R}{3}\ \tan\ \theta_g$

$$= 4{\cdot}2\ R\ \tan \qquad (24)$$

where θ_g represents the blade angle at $\frac{2}{3}$ R.

EXPERIMENTAL MEAN PITCH.—This is the pitch value for which there is zero thrust, that is when the forward component of the lift forces acting on the blades, T_L, is equal and opposite to the drag component parallel to the airscrew axis, T_D. See Fig. 119.

$L \cos \phi = D \sin \phi$, and therefore $T = 0$.

Such a condition could obviously not take place in normal horizontal flight, but is possible during a glide, or dive, with engine on. In practice the airscrew exerts a thrust under most conditions and the axial advance* per revolution is less than the experimental pitch, owing to the backwards velocity imparted to the air, i.e., the backward momentum given to the air (the slipstream) must equal the forward thrust. The difference between the actual translational advance of the airscrew and

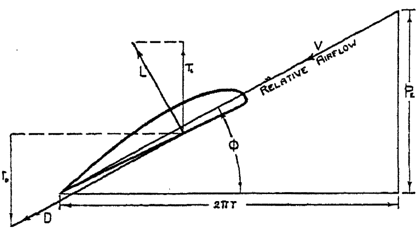


FIG. 119.—EXPERIMENTAL PITCH OF AIRSCREW

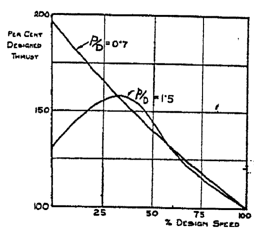


FIG. 120.—VARIATION OF THRUST WITH p/D RATIO

the experimental pitch is called the slip, and is generally expressed as a percentage of the experimental pitch.

Obviously, thrust must be accompanied by slip, and slip is a maximum (100%) when the aircraft is stationary on the ground with the airscrew turning, and decreases as the forward speed of the aircraft increases, depending on both forward speed and engine speed, until it entirely disappears when thrust becomes zero.

Experiments have demonstrated that propellers of large pitch, relative to the diameter, that is of high $\frac{p}{D}$ ratio, give greater efficiency than low pitch values, and are therefore desirable in practice. See Fig. 120. Actually, the choice of pitch is generally restricted by other requirements obtaining for any particular

* Advance here relates to the forward movement of the airscrew into the air, excluding the rearward movement of the air towards the airscrew. In other words, it is the distance in calm air that the airscrew moves relative to the ground.

design of aircraft, such as engine speed, engine torque, and probably permissible airscrew diameter.

Airscrews of low $\frac{p}{D}$ values develop high thrust at low speeds. The blades of airscrews having a pitch greater than 1·2 D become stalled at low forward speeds, with a consequent falling off of thrust at the lower end of the speed range. The stalling results also in decreased r.p.m. due to the greater resistance torque, with further loss of thrust. This is a factor of great importance during take-off.

The curves of Fig. 120 show, in terms of the designed thrust, the variation of thrust, at full throttle with forward speed for airscrews above and below the critical $\frac{p}{D}$ value of 1·2. The loss of thrust with the higher pitch, due to blade stalling, is clearly seen.

Variable Pitch Airscrews.

The variable factors during flight, with which an airscrew has to contend, are flight speed, engine speed and height. It has been seen that the effective angle of attack changes with alteration of engine and flight speeds, and that maximum efficiency for any given diameter of airscrew can take place at one value of $\frac{V}{n}$ only. This means that loss of efficiency is experienced over the greater part of the flight speed range.

Fig. 121, (p. 165), shows that at low speed a small $\frac{p}{D}$ ratio gives the higher efficiency, η, whilst at high speed the reverse is true. The greater speed range of modern aircraft has also introduced the loss of thrust power, due to blade stalling, at the lower forward speeds. This is brought out by the two curves of Fig. 120 (p. 162), in which the lower pitch value, 0·7 D, corresponds to a top speed of roughly 110 m.p.h., whilst the 1·5 D pitch airscrew would be fitted to an aircraft of 250 m.p.h. top speed.

The need for a variable pitch is clearly seen, and further. the need increases with increase of top speed. There would be little gain by fitting a variable pitch propeller to slow speed aircraft, as may be gathered from an inspection of the efficiency

curves in Fig. 121. Only when blade stalling at low speed manifests itself, that is when the pitch exceeds the diameter, such as would be used with an aeroplane having a top speed of over 150 m.p.h., is a variable pitch worth while.

The boosting of engines to increase r.p.m. and power during the take-off is not of much use, since the blades, being already stalled, become still further stalled, and only a small proportion of the additional engine power is converted into thrust power. The loss of thrust caused by the stalled blade is far more serious than the loss of engine speed.

Similarly the use of a two-speed gear for the engine is of little advantage with a fixed pitch airscrew, for although the gear enables the engine speed and hence power, to be increased when the airscrew rate of rotation is low, the slight increase in thrust is offset by the increased tendency of the blades to stall.

The employment of variable, or controlled, pitch enables the use of full engine speed and power at all flight speeds : Efficiency is also increased, since the angle of attack of the blades may be decreased until full power is developed.

For top speeds between 140 and 200 m.p.h., a variable pitch airscrew is very desirable. At still higher speeds a two-speed gear for the engine is also desirable.

Both climb and ceiling are improved by the aid of the variable pitch mechanism, the take-off run is reduced, and the performance at height is improved.

The method of operation of variable pitch airscrews is dealt with in Chapter XII, Volume II.

Airscrew Efficiency.

The efficiency of an airscrew is determined by the ratio of useful work done in propelling the aircraft, to the work supplied by the engine. Thus the efficiency is zero when the aircraft is stationary, for, although the airscrew is exerting some force on the member, or members, restraining its forward progression, no useful work is being done. And again, when the distance moved forward during one revolution of the airscrew is equal to the experimental pitch (point C, in Fig. 117), no propulsive force is being obtained from the airscrew, and again, efficiency is nil. This takes place, of course, at a value of $\frac{V}{nD}$ for which the thrust coefficient, C_T, is zero. (Fig. 116.)

Between these two conditions the airscrew efficiency varies, the average maximum value for most airscrews being about 80 per cent. at a $\frac{V}{n\,D}$ value of roughly $\frac{2}{3}$ to $\frac{3}{4}$ of the $\frac{V}{n\,D}$ range as determined above. In certain instances the efficiency figure has approached 90 per cent.

The work done per second in moving an aircraft through the air at a velocity V ft. per second is T V ft. lb. Also the work done by the engine is $2\,\pi\,n\,Q$ ft. lb. per sec.

$$\text{Hence the efficiency, } \eta = \frac{T\,V}{2\,\pi\,n\,Q} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (25)$$

or inserting the values of T and Q,

$$= \frac{C_T \frac{\rho}{2}\,n^2\,D^4\,V}{2\,\pi\,n\,C_Q \frac{\rho}{2}\,n^2\,D^5}$$

$$= \frac{C_T}{2\,\pi\,C_Q} \times \frac{V}{n\,D} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (26)$$

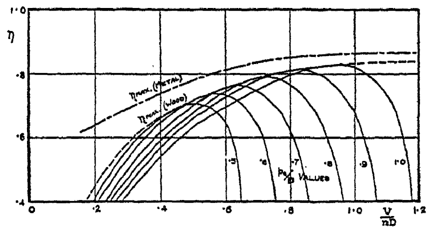


FIG. 121.—AIRSCREW EFFICIENCY CURVES

Fig. 121 shows efficiency curves for wooden airscrews of varying $\frac{p_g}{D}$ values, together with the envelope curve of maximum efficiency for airscrews plotted against $\frac{V}{n\,D}$. Metal airscrews are generally more efficient than wooden airscrews, particularly for high speed work where the relatively thick blades of wooden airscrews are more affected by compressibility of the air. The increased efficiency of metal screws is roughly 5 per cent. at $\frac{V}{n\,D}$ of 1·2, rising to 15 per cent. at 0·4, as shown by the chain-dotted curve of Fig. 121.

Effect of Height on Airscrew.

Owing to the decrease in air density with height, there is a falling-off of engine power, accompanied by a reduction of the resistance met with by the airscrew, which generally results in a loss of power, but a slight increase in the airscrew efficiency. The variation of engine power with altitude is approximately proportional to $\rho^{1.2}$ whereas the power consumption of an airscrew is proportional to ρ. Supercharging may be resorted to in order to increase the charge in the engine cylinders, and so maintain the power at any particular height as desired. Owing, however, to the rarified condition of the air in which the airscrew acts, there is a tendency for it to race and exceed the maximum allowable r.p.m., in which case recourse to throttling becomes necessary.

In this way the consumption of fuel is economised, but it is noticeable that the full available engine power is not utilised. In some cases the airscrew is designed to absorb the full engine power at some predetermined height, but only at the expense of a loss of r.p.m., and therefore power, at the take-off and lower altitudes, unless a variable pitch propeller is employed.

Interference Effect of Fuselage, or Engine Nacelle.

Contrary to what might be expected, tests have shown a slight increase in thrust and airscrew efficiency when working just in front, or behind a body. This is due to the retardation of the air velocity as it approaches the plane of the propeller and also to the fact that air is directed outwards to the more effective parts of the blades. The experimental pitch also is increased. If, however, the additional drag of the body behind the airscrew is taken into account, there is of course a nett loss in thrust.

Effect of Airscrew Speed.

When the speed of an aerofoil relative to the air approaches the speed of sound (1,118 ft. per sec.), the type of flow undergoes a change, and compression of the air takes place. Actual tests have shown that, above a speed of about two-thirds the rate of sound travel, the C_L values commence to decrease, with corresponding increases of C_D. Thus at high propeller speeds

there is a falling off of thrust, with increased torque values, these effects being obviously more noticeable near the tips where the speed is highest and where the thrust is greatest.

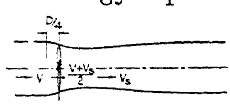
In order to avoid the ill effects of high speed, the rate of revolution should be kept down to give a tip speed not exceeding about 750 ft. per sec., for wooden airscrews, but may be increased up to 1,000 ft. per sec. in the case of thin metal blades. Thus for a 10 ft. diameter wooden propeller, the maximum r.p.s., would be $n = \dfrac{750}{\pi \times 10} = 24$ approximately, or 1,420 r.p.m. Here again, the advantage of gearing is obvious.

Effect of Slipstream.

The slipstream effects may be divided into that due to mutual interference between airscrew and body and the interference-effect between the airscrew and main lifting surface.

Dealing first with the airscrew/body effects, the air passing through the propeller disc forms a cylinder, of diameter roughly 95 per cent. of that of the propeller, which receives a backwards velocity together with a rotational motion. The backwards component *added* to the forward speed of the aircraft gives V_S, $(V + v = V_S)$, the translational velocity of the air relative to those parts of the aircraft within the slipstream cylinder. The slipstream velocity is given approximately by the formula $\dfrac{V_S}{V} = \sqrt{1 + \dfrac{0{\cdot}11\,\eta}{\left(\dfrac{V}{nD}\right)^2}}$ and may be anything from 3 to 30 per cent. greater than V at η_{max}, increasing rapidly with decreased speed of flight.

The drag of all parts within the slipstream cylinder is increased in the proportion $\left(\dfrac{V_S}{V}\right)^2$. It is equivalent to a loss of thrust. There is another slipstream drag effect due to the horizontal velocity gradient behind the airscrew. Due to the energy imparted by the blades, the pressure rises immediately behind the airscrew disc. This pressure converts into kinetic energy, with increase of velocity (see Fig. 122), and there is therefore a falling pressure



IG. 122.—Slipstream Velocity Diagram

away from the airscrew. This adds another component to the total drag and cannot be neglected. But, as already explained, the presence of a body behind the airscrew has the effect, generally, of increasing the efficiency of the latter. A nacelle, or fuselage, set approximately on the airscrew axis diverts the air from the central, inefficient region to the more effective, outer parts of the blades. These two aspects roughly cancel out, and if this may be assumed without serious error, then the airscrew/body effect may be simply dealt with by using the efficiency curve for the free propeller, and multiplying the drag of all parts coming within the 0·95 D cylinder by $\left(\frac{V_S}{V}\right)^2$, where V_S is calculated from the given formula.

In the case of a pusher airscrew, the presence of the body in front also improves the airscrew efficiency, but the change of drag is not so pronounced as with the tractor screw. The speeding up of the flow due to the airscrew commences at a region several diameters in front of the airscrew and becomes perceptible at a section distant $\frac{D}{4}$ from the plane of the screw, and at the propeller disc the air has received about one-half of the velocity increase, the speed then being $\frac{V + V_S}{2}$. There is, however, a drag increment due to horizontal pressure gradient as with the tractor screw. It is fairly safe to assume that one-half of the body is subjected to a relative air-flow of velocity V and the other half to the slipstream velocity V_s, without alteration of the free airscrew efficiency. The drag of parts within the slipstream cylinder may then be multiplied by the factor $\frac{1 + \left(\frac{V_S}{V}\right)^2}{2}$

Coming now to the effects of the slipstream on the main lifting surface, the increased air speed gives an increased lift, so that the aircraft may be flown at a smaller angle of attack than would be the case of a wing surface unaffected by the slipstream, and this results in some decrease of the total drag. The most important feature of this latter effect is the improvement in the take-off qualities of an aircraft, and a lowering of the minimum speed of flight with engine on, especially since the slip, or slipstream velocity, is greatest at low forward speeds of the aircraft. The slipstream over a wing also acts as a deter-

rent to stalling at large angles of incidence. The power of the tail control surfaces is also improved when within the slipstream area, a particular advantage again at low flight speeds with engine on.

There is still the rotational velocity of the slipstream to be considered. This acts in the same direction as the propeller rotation, and sets up air loads on all tail surfaces, and, to a minor extent, on the fuselage sides. The loads thus formed act upward on one side of the tail-plane and down on the other, which produces a twisting effect about the longitudinal axis. The vertical tail fin is often offset slightly so as to lie along, or beyond, the air-flow path, and thus avoid the turning effect that would otherwise result.

It has already been mentioned that the airscrew torque tends to rotate the aircraft in the direction opposite to that of its own rotation, and since the slipstream twisting effect acts in the same sense as the airscrew, the slipstream effect tends to diminish the torque effect to some extent.

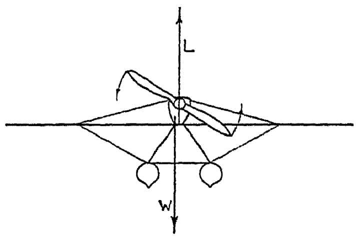


FIG. 123.—AIRSCREW TORQUE EFFECT

Torque effect is of chief importance with small, comparatively light aeroplanes fitted with powerful engines, particularly when the span is short, with a consequent small moment of inertia about the longitudinal axis. In some cases the effect is overcome by wash-out of incidence over the wing which would otherwise tend to rise, i.e., the side on which the airscrew descends, whilst another method is to arrange for the centre of gravity of the aircraft to be on one side of the centre of lift, and so set up a couple tending to rotate the machine in the direction of the airscrew rotation. This has been accomplished in the case of seaplanes by the use of a larger float on one side, inside which fuel is carried. (Fig. 123.)

Gyroscopic Effect.

One further effect of the rotating airscrew that may receive consideration is the gyroscopic effect. The revolving airscrew has the well-known quality of the gyroscope, that is, a strong tendency to resist any change of plane of rotation. This becomes noticeable chiefly during turns made by the aircraft, and causes the nose to rise, or fall, according to the direction of engine rotation and the direction of turn. Thus, if the turn has the same sense as the airscrew rotation, say clockwise, the tendency is for the nose to fall, and vice versa.

CHAPTER XIII

AEROPLANE PERFORMANCE

FIGURES for the probable performance of an aircraft are necessary during the design stage, in order to ascertain its degree of usefulness for various purposes, and for comparison with any stipulated minimum criteria as regards top speed, climb, landing-speed, ceiling, etc. Likewise the effects on performance of the several determining factors should be known if a machine is to be designed to fulfil a specific purpose.

Horse Power Available and Required.

The engine power is obtained from the b.h.p./r.p.m. curve for the engine employed (Fig. 110, Chap. XII).

It now remains to select a number of speeds, say at 10 m.p.h. intervals, from below stalling speed to above the estimated top speed, and to find the r.p.m., and hence b.h.p., and the nett airscrew efficiency for each speed. In this way values of power available are found and the curve P_a is plotted against V.

The resistance to forward motion of an aircraft is considered as made up of two parts; that due to the wings, and that due to the remainder, known as parasite drag.

Parasite Drag.

The body drag varies approximately as the square of the flight speed, and its importance therefore is greatest at high speeds of flight. The change of attitude at which the fuselage is presented to the relative air-flow causes some modification of the V^2 proportion, but since the total range of angles is small, perhaps 10° either side of normal, and the fuselage is long in comparison with its width, i.e., the fineness ratio is large, the effect of small changes of attitude is not of great importance, and is probably rendered less so on account of the down-wash from the wings.

The parasite resistance of an aircraft can be estimated from wind tunnel tests on models, or by comparison with known resistances

of similar shapes from which $C_{D'}$ values may be obtained. The body, or parasite, drag, D′, can then be calculated, due allowance being made for interference effects.*

Drag tables should be prepared for different speeds, taking account of the values of R.N. for each component at the various speeds, and also the inclination of the various components, including that of the tail-plane required for trim.

The components of drag within and without the slipstream, dealt with in the previous chapter, should be calculated separately —the slipstream diameter being taken as 0·95 D.

Wing Drag.

The drag due to the wings can be obtained from the characteristic curves for the wing section employed, corrected, if necessary, for the aspect ratio adopted. In the case of a biplane, modification for "biplane effect" is also essential.

Values of C_L for various flight speeds may be calculated, and the corresponding C_D figures, i.e., at the same incidence values, can be applied to the formula

$$D = C_D \frac{\rho}{2} SV^2 \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (27)$$

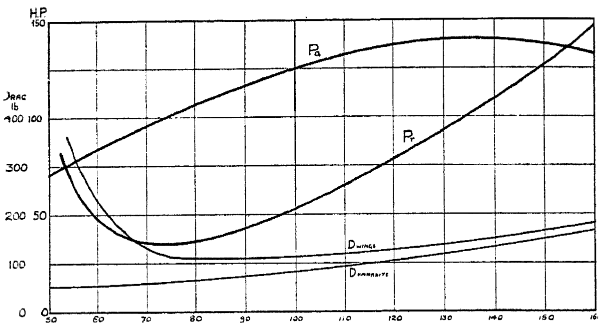


FIG. 124.—CURVES OF DRAG, POWER REQUIRED AND POWER AVAILABLE

Fig. 124 shows the wing and body drags plotted on a speed base, together with the curve for power required, P_r, which is obtained from the formula

$$P_r = \frac{D_t V}{550} \quad . \quad . \quad . \quad (28)$$

where $D_t = D + D'$.

* See Chap. XIV.

From the curve for P_r it is seen that there is one speed at which the power required is a minimum, above and below which greater power is necessary.

The total drag is a minimum at a speed roughly 30 per cent. above stalling, and the power required is minimum at about 5 to 10 per cent. above stalling speed. This latter point is of interest in the consideration of take-off, as it obviously offers the best conditions for getting a heavily loaded aeroplane off the ground.

Speed Range, Horizontal Flight.

The intersections of the P_a and P_r curves of Fig. 124 give the approximate minimum and maximum speeds at which level flight is possible with fully opened throttle.

Between these values there is an excess of available power over that required for level flight, which may be used for climbing.

Minimum Horizontal Flight Speed.

The lowest speed for horizontal flight is attained when the wings are at their position of $C_{L\,max}$, the speed then being

$$V_{min} = \sqrt{\frac{W}{C_{L\,max}\frac{\rho}{2}S}} \quad . \quad . \quad . \quad . \quad . \quad (29)$$

An allowance may be made for slipstream velocity over the affected part of the wing, which reduces the minimum flight speed to some small extent. In this connection it may be mentioned that the engine thrust line in some cases is down-set to increase the effective wing incidence and thereby assist conditions of take-off.

If, however, as is usual, the thrust is inclined upwards at some angle when the aircraft is at the slow speed attitude, then there is a component of thrust helping to support the weight, or $L + T\sin\theta = W$,

$$\text{whence } V_{min} = \sqrt{\frac{W - T\sin\theta}{C_{L\,max}\frac{\rho}{2}S}} \quad . \quad . \quad . \quad . \quad (30)$$

A formula for rapid prediction of minimum speed, of use with average present-day aerofoils, is

$$V_{min} = 17\sqrt{\frac{W}{S}} \quad . \quad . \quad . \quad . \quad . \quad . \quad (31)$$

Maximum Horizontal Flight Speed.

$$T\ V = \eta\ P\ 550, \text{ where } \eta = \text{airscrew efficiency,}$$

$$\text{but } T = D_t$$

$$\text{hence } \eta\ P\ 550 = C_{Dt}\frac{\rho}{2}SV^3$$

$$\text{or } V_{max} = \sqrt[3]{\frac{\eta\ P\ 1100}{C_{Dt}\ \rho\ S}} \text{ ft. per sec.} \quad . \quad . \quad . \quad (32)$$

The Everling High Speed formula given in an earlier chapter is obtained from this formula by ascribing suitable values to η and C_{Dt} for various types of aircraft. For rapid calculation, in the initial stages of a design, C_{Dt} is taken as $C_{D\ min}$ for the section employed, to which is added the parasite drag increment; η is taken as 0·8, and P is the maximum available h.p.

A rough and ready formula for top speed, based on (32), is

$$V_{max} = K\sqrt[3]{\frac{P}{S}} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (33)$$

where K varies from 140 to 180, with the value of 160 for an aerodynamically clean single-engined aircraft, and 170 for a clean two-engined machine.

Rate of Climb.

The vertical distance separating the curves of P_r and P_a (Fig. 124) shows the excess of power, P_c, which may be used for altitude increase, and the best climbing speed is that for which the excess power is a maximum.

The rate of climb is then :

$$V_c = \frac{P_c \times 33000}{W} \text{ ft. /min.} \quad . \quad . \quad . \quad . \quad . \quad (34)$$

$$\text{where } P_c = P_a - P_r.$$

The optimum speed for climb is roughly one-third of the speed range greater than minimum speed.

The angle of climb, θ, is given by

$$\sin \theta = \frac{V_c}{V} \qquad . \quad . \quad . \quad . \quad . \quad . \quad . \qquad (35)$$

Care should be taken to see that both velocities have a common scale of units.

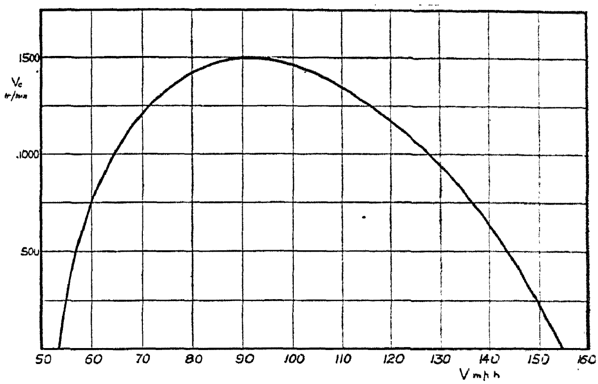


FIG. 125.—RATE OF CLIMB

Gliding Flight.

In gliding flight (see Fig. 126)

$$W = \sqrt{L^2 + D_t^2}.$$

The angle of glide, θ, is given by

$$\tan \theta = \frac{D_t}{L} \qquad . \quad . \quad . \qquad (36)$$

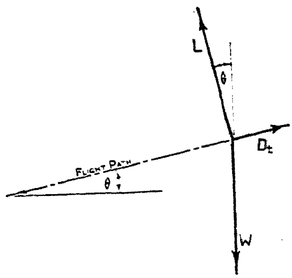


FIG. 126.—GLIDING FLIGHT

Note that D_t is total drag and that L* is less than W.

The minimum angle of glide occurs at a speed and incidence for which $\frac{L}{D_t}$ is a maximum, and is the condition for covering the greatest horizontal distance for a given loss of height.

* This is the nett lift and probably includes a down load on the tail so that the wing lift may not be less than W.

The speed in gliding flight is

$$V = \sqrt{\frac{W}{\sqrt{C_L^2 + C_{Dt}^2} \times \frac{\rho}{2} S}} \quad . \quad . \quad (37)$$

and is a minimum when $(C_L^2 + C_{Dt}^2)$ is maximum.

It is seen that in gliding flight the drag force has a small vertical component opposing the weight.

Effects of Aircraft Weight Variation.

Cases occur in which an aircraft, when built, has a different all-up weight to the weight as originally estimated, whilst in other cases the effects on performance of increased, or decreased, load may be required to be found.

If the weight is changed from W to W_1, let V and V_1 be the corresponding speeds at given values of incidence and lift coefficient.

$$\text{Then } C_L \frac{\rho}{2} SV^2 = W$$

$$\text{and } C_L \frac{\rho}{2} SV_1^2 = W_1$$

$$\text{Whence } \frac{V_1}{V} = \sqrt{\frac{W_1}{W}}$$

C_{Dt} remains unchanged for the given incidence, hence

$$\frac{D_1}{D} = \frac{V_1^2}{V^2} = \frac{W_1}{W}$$

$$\text{and } \frac{D_1V_1}{DV} = \frac{W_1}{W} \times \sqrt{\frac{W_1}{W}} = \left(\frac{W_1}{W}\right)^{\frac{3}{2}}$$

If P_1 is the new value of required h.p., then

$$P_1 = P\left(\frac{W_1}{W}\right)^{\frac{3}{2}} \quad . \quad . \quad . \quad . \quad . \quad (38)$$

By the aid of this formula values of the modified h.p. required may be calculated over a range of speed, from which the new power-required curve may be plotted.

Increased all-up weight causes the following main effects :—

(*a*) Maximum horizontal speed is slightly reduced.

(*b*) Minimum horizontal speed is increased in proportion to the square root of the weight increase, $V_1 = V\sqrt{\frac{W_1}{W}}$, and

(*c*) Rate of climb is considerably reduced.

The small loss of top speed with increased weight is due to the slow rate of C_{Dt} increase at the smaller incidences of top speed flight. Loss of rate of climb is much more marked owing to the greater rate of C_D increase at the higher incidences required for climb, together with the extra weight that has to be lifted.

As an example of the effects of change of weight, consider an aeroplane of normal weight, 3,000 lb., requiring 120 h.p. at a speed of 100 m.p.h., which carries an overload of 500 lb.

For flight at the same incidence, the new speed will be

$$V_1 = V\sqrt{\frac{W_1}{W}} = 100 \times \sqrt{\frac{3500}{3000}} = 108 \text{ m.p.h.}$$

$$\text{and } P_1 = P\left(\frac{W_1}{W}\right)^{\frac{3}{2}} = 120\left(\frac{3500}{3000}\right)^{\frac{3}{2}} = 151{\cdot}4 \text{ h.p.}$$

In this way a series of values of power required may be calculated from which the new curve can be drawn.

Wing-Loading.

By wing-loading is meant the total all-up weight of the aircraft divided by the wing area, $\frac{W}{S}$, or the gross weight carried by each square foot of wing surface, assuming an even distribution. The formula for minimum speed, formula (29), may be written

$$V_{\text{min}} = \sqrt{\frac{1}{C_{L\max}\frac{\rho}{2}}} \times \sqrt{\frac{W}{S}} \text{ ft. per sec.}$$

From which it is immediately apparent that *a low value of wing-loading signifies a low minimum or landing speed.* Thus an increase of wing loading from 10 to 20 lb. per sq. ft. would increase the minimum speed by $\frac{\sqrt{20}}{/} = 1{\cdot}41$, say.

On the other hand, formula (32) shows that maximum speed is proportional to $\sqrt[3]{\frac{1}{S}}$.

In other words a small wing area, with consequent *high wing loading, is conducive to high speed.* Actually this assumes that C_D remains unaltered, whereas a reduction in S must result in increased C_D, due both to the larger incidence required for the smaller wing, and to the fact that the parasite drag component $C_{D'}$ is inversely proportional to the wing area.

The rate of climb has been given as

$$V_c = \frac{P_c\ 33000}{W} \text{ ft. per min.}$$

$$\text{or } \frac{(P_a - P_r)\ 550}{W} \text{ ft. per sec.}$$

But $P_r = \dfrac{D_t V}{550}$. Hence

$$V_c = \frac{550\ P_a}{W} - \frac{D_t\ V}{W} \text{ ft. per sec.} \quad \ldots \ldots \quad (39)$$

For a high rate of climb the value $\dfrac{D_t V}{W}$ should be low.

$$\text{Now } \frac{D_t V}{W} = \frac{D_t V}{L} = \frac{C_{Dt}\frac{\rho}{2} SV^2}{C_L \frac{\rho}{2} SV^2} \sqrt{\frac{W}{C_L \frac{\rho}{2} S}}$$

$$= \sqrt{\frac{W}{S}} \times \frac{C_{Dt}}{C_L^{1 \cdot 5}\sqrt{\frac{\rho}{2}}} \quad \ldots \ldots \ldots \quad (40)$$

from which it is seen that *low values of wing-loading and total drag are beneficial for climb,* though to some extent these factors are antagonistic.

Summing up, then, the wing-loading should be high for high top speed, but should be kept low for low landing speed and a high rate of climb.

Power-Loading.

The all-up weight divided by the engine horse-power gives the power-loading, $\dfrac{W}{P}$. Top speed has been seen to be proportional (Formula 32) to $3\sqrt{\dfrac{P}{S}}$, and therefore to $\dfrac{P}{W}$, which shows that

a low power-loading is consistent with a high top speed. As explained under the heading "Wing-loading," the change of the C_D value, consequent on change of S, causes some modification of the proportion given above.

The effect of power-loading on minimum speed is not generally of great consequence, since it is the landing, or engine-off, condition that is of most importance. It was seen, however, that the engine thrust helps to support the weight, and therefore of two aircraft having equal wing-loading, the one with the lower power-loading will have the slower minimum speed in horizontal flight.

For considerations of climb, the effects of power-loading are far more pronounced. Formula (39) indicates at once that *rate of climb is roughly inversely proportional to the power-loading.*

Power-loading should therefore be low for considerations of top speed and for rapid climb.

Span Loading.

Span loading, or more strictly span squared loading, $\frac{W}{b^2}$, is given as the total weight divided by the square of the span, and forms a basis for comparison of aircraft performance, particularly at the higher values of C_L, i.e., at the take-off and climb. The importance of aspect ratio has already been referred to, where it was seen that induced drag, of most importance at the higher incidences, varies inversely as the aspect ratio, or in other words a high aspect ratio means relatively low induced drag. Now it has been seen also that a low value of wing loading is beneficial for flight at the higher values of C_L.

Taking these two qualities together it is obvious that the performance of an aeroplane at coarse incidences depends on a low figure for span squared loading, for

$$\frac{\text{wing loading}}{\text{aspect ratio}} = \frac{\frac{W}{S}}{\frac{b^2}{S}} = \frac{W}{b^2}$$

Furthermore the disadvantages of a high wing-loading may be compensated for, to some extent, by the employment of a high aspect ratio.

The induced drag, referred to above, has been shown as

$C_{Di} = \frac{C_L{}^2}{\pi A}$. The relation between induced drag and weight is therefore

$$\frac{D_i}{W} = \frac{C_{Di}}{C_L} = \frac{C_L{}^2}{\pi A C_L} = \frac{C_L S}{\pi b^2}$$

$$= \frac{C_L \frac{\rho}{2} S V^2}{\pi b^2 \frac{\rho}{2} V^2}$$

$$= \frac{W}{b^2} \times \frac{1}{\pi \frac{\rho}{2} V^2} \quad \ldots \ldots \quad (41)$$

Hence, for a given weight and speed, the induced drag is proportional to the span² loading. The figure for span² loading of present-day aeroplanes varies from 8 for racing aircraft, where high main-plane incidence is of little importance, to 2 for light aeroplanes and freight-carrying aircraft, to less than 1·0 for ultra-light aeroplanes and even lower than 0·15 in the case of some high-efficiency sailplanes.

Effects of Height.

So far performance has been considered for sea-level, or normal atmospheric, conditions. The air density decreases with altitude, which results both in a loss of engine power and in a reduction of air drag, so that the P_a and P_r curves of Fig. 124 are both altered for any altitude other than sea-level.

Dealing first with the alteration of engine power, it is obvious that a decreased density means a reduced weight of charge supplied to the engine cylinders, unless supercharging is resorted to, and hence less power is developed. It would appear, then that the engine power varies directly as density : This, however, is not quite correct owing to other factors, chief of which is the fact that frictional losses of the engine do not fall off with density, whilst at the same time loss of engine speed may cause a reduction of airscrew efficiency. A formula which takes into account, with reasonable accuracy, the influences of density, airscrew efficiency, etc., on horse power available is

$$P_{aH} = P_a \left(\frac{\rho_H}{\rho}\right)^{1.4} \quad \ldots \ldots \quad (42)$$

where P_{aH} and ρ_H refer respectively to power and density at the height considered.

Thus, for example, the density at 10,000 ft. in lb. per cu. ft. is 0·056 against 0·077 at sea-level. Hence the engine power at that height would be

$$P_{aH} = P_a\left(\frac{\cdot 056}{\cdot 077}\right)^{1.4} = 0\cdot 6412P_a,$$

or say 64% per cent. of the sea-level power. In this way a new curve for power available may be constructed, assuming of course that the loss is not made good by supercharging.

The next consideration is the power required.

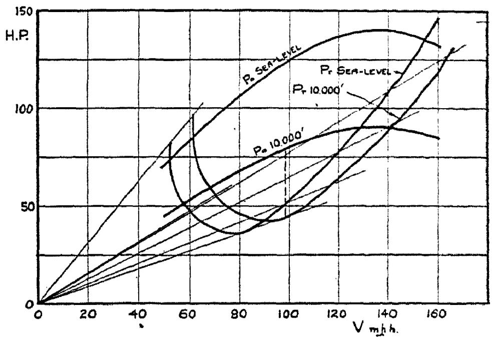


FIG. 127—EFFECT OF HEIGHT ON PERFORMANCE

At a given speed, reduced air density must be compensated for by an increase of the angle of incidence.

There is an increase in induced drag but a decrease in parasite (including profile) drag, the former constituting but a small proportion of the latter, so that there is a nett reduction of the power required.

If the attitude of the aircraft is kept unchanged at the higher altitude there must be an increase in speed to provide the necessary lift, i.e., for the lift to remain constant, and induced drag will therefore be unaltered, as also will the total drag. The change in speed is inversely proportional to the square root of the air density, the power required being equal to the product of drag and speed, or since the drag remains constant, the power must be proportional to the square root of the air density.

For constant incidence and lift then, the power required at the greater speed is given by

$$P_{rH} = \sqrt{\frac{\rho}{\rho_H}}\, P_r \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (43)$$

whilst the speed has increased in the same proportion.

The modified curve may be obtained by calculating a series of values of P_{rH} and V, or more simply by drawing a series of lines radiating from the origin of the chart ($V = 0$ and $P = 0$) to cut the sea-level P_r curve at a number of points. The revised curve is then obtained by marking points on these lines distant from the origin $\sqrt{\frac{\rho}{\rho_H}}$ times the distance of the original intersection points from the origin. Thus for an altitude of 10,000 ft.

$$\sqrt{\frac{\rho}{\rho_H}} = \sqrt{\frac{\cdot 077}{\cdot 056}} = 1{\cdot}173,$$

and at the new curve obtained by this method is shown in Fig. 127.

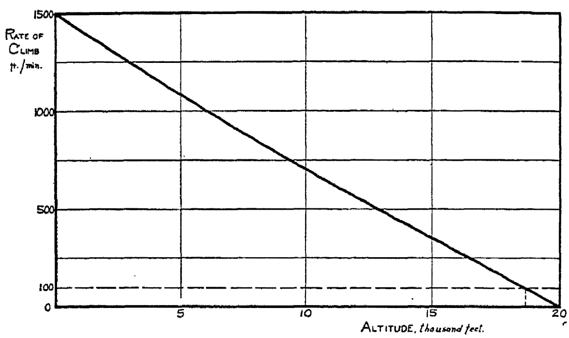


FIG. 128.—CLIMB AND HEIGHT

The intersections of the new P_a and P_r curves give the minimum and maximum horizontal flight speeds at the greater altitude, the result being an increase in the former and a decrease in the latter, with a considerable lowering of the speed range.

The rate of climb is also adversely affected.

Ceiling.

The rate of climb may be calculated for a series of heights

and plotted against height. The altitude at which the rate of climb falls to zero is the *absolute ceiling*, whilst the *service ceiling* is defined as the height at which the maximum rate of climb is 100 ft. per minute, and is easily found from the rate of climb curve.

Ground Effect on Performance.

The effects of the close proximity of ground, or water, to the wing of an aeroplane are generally beneficial, resulting both in increased C_L and in reduction of C_D with considerable improvement of $\frac{C_L}{C_D}$ max.

Ground effects have been studied in the wind tunnel (*a*) with an imitation ground, stationary relative to the aerofoil, and (*b*) by the reflection method, in which a second aerofoil is mounted below the wing being studied, and in the inverted position, so that the air-flow mid-way between the two aerofoils has a horizontal flow and thus represents the presence of the ground.

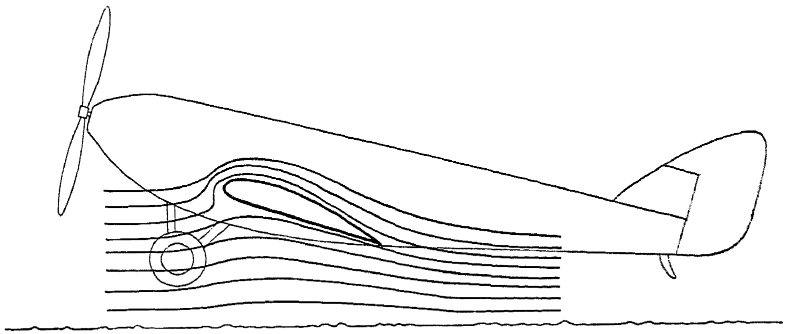


FIG. 129.—AIR-FLOW PAST WING NEAR GROUND

The former method is faulty since the ground should move with the relative wind. The second method is correct, for the speed of the imaginary plane is approximately the same as that of the free air and thus provides a condition similar to flight near the ground.

The chief effect of ground proximity is the reduction in induced drag, explained by the restricted downwash angle. This reduction amounts to about 5 per cent. at a height equal

to the span, increasing to about 40 per cent. when the gap equals the chord, 60 per cent. at one-half chord height, and 70 per cent. at one-quarter chord.

At take-off a reduction of 40 per cent. may mean a decrease of 20 to 25 per cent. in the total drag, whilst a top speed increase of from 1 to 3 per cent. may be obtained close to the ground.

The improvement in lift is probably caused by the lowered pressure at the trailing-edge, due both to the increased venturi effect and the greater relative speed of the air-flow past the wing resulting from ground friction. This low pressure region at the trailing-edge helps to maintain smooth flow over the upper surface. The greater relative air speed, already referred to, or the retarding effect of the ground upon the mass of air being dragged forward by the aeroplane, is also likely to be a factor contributing to improved lift.

The increase in $C_{L\ max}$ is roughly 2 to 3 per cent. at a height of one-half chord above the ground and as much as 10 per cent. at one-quarter chord, the effect being still more pronounced at medium angles of incidence.

This may result in a lower possible take-off speed and in reduction of perhaps 1 to 3 per cent. in the landing speed.

The drag effects are most pronounced, as might be expected, with wings of low aspect ratio.

CHAPTER XIV

PARASITE DRAG AND INTERFERENCE DRAG

PARASITE drag has received a certain amount of attention in the earlier chapters, but since it forms a special study in itself it has been made the subject of a separate chapter which is included here for more detailed examination.

Parasite Drag.

Strictly speaking, parasite drag is that component of the total resistance of an aircraft due to parts that do not contribute lift. Although the fuselage, and other parts for that matter, do generally augment the lift to some extent, it is nevertheless regarded as constituting a parasite drag member.

As has already been stated, profile drag consists of two components, skin friction and form drag, the proportion of each depending on the body shape. The latter is comprised of eddy formation and " pressure " drag due to the horizontal resultant of pressure distribution over a body: This is, however, very small in well-shaped forms and is generally neglected.

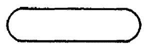
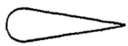
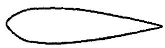
Simple Geometric Shape.

Discussions on parasite drag generally commence with the flat plate held perpendicular to the air-flow as representing both the least streamlined shape and the case in which the eddy resistance component assumes the greatest dimensions, the skin-friction component being in fact almost negligible. C_R may be taken as 1·2, and there is little difference between the values for circular and square plates of similar area, but elongation of one axis of the plate tends to increase C_R. The value also tends to increase with Reynolds' number.

Increasing the thickness of a circular disc, i.e., its conversion to a cylinder with its axis parallel to the direction of flow, allows some straightening of flow between the two ends with consequent

reduction of resistance. Considerably greater drag reduction accompanies the conversion of a flat plate into a sphere, the coefficient being about 0·12, or but one-tenth that for the plate. The insertion of a cylindrical portion at the maximum section of a sphere (see Table IV) does not have quite the beneficial effect that might be expected and by increasing the length of the cylinder, the additional skin friction may more than compensate for the reduction of eddy resistance.

TABLE IV.—RESISTANCE OF VARIOUS BODIES

Shape		C_D	Index
Flat Plate		1·2	100
Cylinder, Axis Parallel to Flow ..		0·9	75
Sphere		0·12	10
Cylinder, Capped with Hemispheres ..		0·10	8
Hemisphere faired with Cone ..		0·06	5
Streamline Shape, Fineness Ratio 4·5		0·03	2·5
Streamline Shape, Fineness Ratio 2·5		0·026	2·2

For the shapes dealt with so far the drag component due to eddy formation is highly predominant, at least 90 per cent. of the total.

The next shape, obtained by fairing a hemisphere with a conical tail, may be regarded as a simple streamline form, and has a resistance roughly half that for the previous two bodies, or but one-twentieth of the flat plate drag. In the reversed position, i.e., with the thin end forward, the resistance is nearly doubled.

Streamline Shapes.

The drag of a good streamline shape consists of 75 per cent., or even more, skin friction and is little more than one-fiftieth that of the flat plate. Since frictional drag plays so large a part, an increase of fineness ratio results in greater resistance. On the other hand as the ratio is decreased below 2 the growth

of eddy resistance outweighs the saving on skin friction. A fineness ratio of from 2 to 3 gives best results ; but if the streamline contour is poor a larger ratio of length to diameter may be beneficial.

Apart from the question of fineness ratio the position of maximum diameter is also important and a location at 40 per cent., or rather greater, of the length from the nose is best. Obviously sudden change of contour should be rigorously avoided.

In the case of airship hulls the question of volume is also of primary importance, the criterion volume/drag resulting in fineness ratios rather greater than that for minimum C_R being employed.

Table V gives the co-ordinates for a good streamline shape that is probably little inferior to any that have so far been evolved.

TABLE V.—CO-ORDINATES OF STREAMLINE SHAPE

Distance from Nose	Diameter/Max. Diameter
0	0·0000
0·0125	0·2478
0·025	0·3480
0·05	0·4841
0·1	0·6615
0·2	0·8660
0·3	0·9682
0·4	1·0000
0·5	0·9765
0·6	0·9049
0·7	0·7818
0·8	0·6001
0·9	0·3469
0·95	0·1869
1·00	0·0000

Effect of Reynolds' Number.

So far no mention has been made of the effect of Reynolds' number on the resistance of the shapes considered. In Chapter II mention was made of the change-over from laminar to turbulent flow within the boundary layer.

Fig. 130 depicts the resistance due to skin friction plotted

against Reynolds' number, from which it is noted that separate curves exist for laminar and turbulent boundary layer flow, there being a critical region between the two curves within which the transition from one type of flow to the other takes place. The general trend of each curve is to show decreased resistances with increase of R.N., though where the transition comes into effect the resistance suffers an increase.

It would be expected that in the case of the streamline shape for which drag is largely composed of skin friction, the curve giving resistance against R.N. would be similar to the diagram just considered and this is approximately true. The drag is a minimum at R.N. of roughly 1×10^6, increases thence by about 50 per cent. to R.N. 2×10^6, after which there is again a gradual decrea--

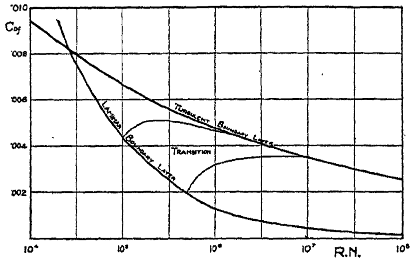


FIG. 130.—SKIN FRICTIONAL DRAG

For shapes other than good streamline, the flow pattern and point of separation become the deciding factors, which in turn have been seen* to depend to some extent on the type of flow within the boundary layer, turbulence in this case being beneficial for low drag. Consequently curves for C_R against R.N. are far more irregular. In general there is a fall of C_R with increasing R.N., but over certain ranges there is rapid increase of resistance with R.N. (Fig. 139).

Surface Roughness.

The degree of surface smoothness of aircraft is of very considerable importance and has received greater attention of late years. The effect of a roughened surface is to induce turbulence

* See p. 21 (Chap. II).

in the boundary layer, and this has been seen to delay separation and so to be beneficial as regards drag due to eddy formation. As the degree of roughness becomes greater the irregularities tend to increase skin friction, so that there is both a gain and a loss. How these two components of drag balance out depends on the conditions. It has already been seen that the boundary layer thickness decreases with velocity and it is only natural that to avoid excessive skin friction surfaces should become smoother with increase of speed.

The deciding factor is now known to be the depth of the sub-layer in the boundary lamina,* and aerodynamic smoothness is said to have been obtained when the depth of the surface roughness is no greater than the thickness of the sub-layer. Experiments† appear to have shown that the drag increase is inappreciable until the degree of surface roughness exceeds twice the value as specified above.

Excluding the effect on wings, a small degree of roughness does not appreciably increase the skin friction of aero surfaces, but such drag assumes greater importance with increasing Reynolds' number.

As would be expected, polished metal surfaces show up best in this respect, though plywood and fabric properly treated and polished are not greatly inferior, particularly at lower Reynolds' numbers. Of greatest interest are the effects of roughness over the surface of aerofoils, there being important changes in both lift and drag with degree of roughness, and here the position of the surface area considered is of almost equal importance to the value of the Reynolds' number.

The results of tests carried out in this country‡ and America§ agree as regards general indications but not quantitively, due no doubt to the different test conditions and to the large scale effect. In both cases it was found that surface roughness tends to keep $C_{L\ max}$ constant with increasing R.N. above a certain value, such value increasing with decrease of surface roughness. Losses of $C_{L\ max}$ of as much as 26 per cent., and even 50 per cent., in the British and American tests respectively, were recorded, but it is doubtful whether the larger value is ever approached

* See p. 22 (Chap. II).

† " Profile Drag," B. Melvill Jones, *R.Ae.S. Journal*, May, 1937.

‡ " Effect of Surface Roughness on Characteristics of Aerofoils," *R. & M.* No. 1708.

§ *N.A.C.A.* Tech. Notes 457 and 495.

in practice. Loss of lift is due almost entirely to surface roughness over the forward one-third of the upper surface, the rear two-thirds and the underside having negligible effect in this respect.

Increases in minimum drag of as much as 50 per cent. have been measured, depending on the surface condition, the effect in this instance not being confined to the same extent to the upper surface.

The following Table is given as a rough guide only of the permissible degree of roughness as a fraction of the chord without appreciable loss of aerodynamic performance.

TABLE VI.—ALLOWABLE AEROFOIL SURFACE ROUGHNESS

R.N. $\times 10^6$	Depth of Excrescences C/1,000,000
1	81
2	44
5	19
10	10
20	5·4
50	2·3
100	1·2

Rivet Heads.

The effect on performance of protruding rivet heads is similar to that of a rough surface, but is not so easy to avoid. A small reduction in top speed is the most serious result and this effect, for the wing at least, may be nearly halved by countersinking the rivet heads over the forward one-third of the upper surface. Whether the gain in speed of roughly 1 per cent. is worth while must depend on the use for which the aircraft is employed.

Obviously the leading-edge region of a wing, including the adjacent portion of the under-surface over which the stagnation point travels from the no-lift to maximum lift angle, should be kept severely free from all obstruction.

Fuselage Shape.

It is unnecessary to state that fuselages should approach as close as possible to the ideal streamline shape. Just as obvious

is the statement that some deviation from this ideal is difficult, if not impossible, to avoid. Nevertheless the importance of this factor cannot be too highly stressed if good performance is to be obtained, and at least there are several features that should be carefully watched if the drag is to be less than four times that of a streamline form. C_R for a cabin fuselage approaching the streamline form should not exceed 0·06 but for open cockpit fuselages this value may be doubled or even greater.

Considerations of stability and control and also housing of contents will not allow the optimum fineness ratio to be employed, whilst economy of weight and engine housing tend to give a forward position of the maximum cross-section. Losses of efficiency increase as the sectional shape departs from the circular, and of course the adverse effects of unavoidable protuberances, and excrescences, are less pronounced if kept away from the nose of the fuselage—in other words, a clean front portion is of the greatest importance.

Owing largely, but not totally, to the helical path of the propeller slipstream, a square section fuselage is considerably inferior to the circular shape as regards drag. If the comparison is based on cross-sectional area the square fuselage shows an increased resistance of one-quarter to one-third. The adverse effect of yaw and pitch during flight is much more pronounced with square section than with round and oval bodies.

In order to improve the ground angle of the aeroplane for landing conditions, i.e., the angle of attack of the main planes when the machine is resting on the ground, the fuselage axis is often curved upwards to the rear so that the fuselage top is roughly parallel to the longitudinal axis. With the somewhat high fineness ratio generally employed for fuselages this would not be serious, but by curving the axis relative to the propeller, or slipstream, axis some increase in drag must result. For the same reason best results are obtained when the airscrew axis forms the axis of symmetry of the fuselage. A fuselage with an up-curved axis gives minimum drag at a small positive angle of pitch so that if the main plane is set relative to the minimum drag axis the increase of ground angle due to upsetting the fuselage tail is roughly only half the amount that might be anticipated. However, the advent of the retractable undercarriage has enabled the desired ground angle to be obtained without deformation of the main axis.

Hulls and Floats.

The hull of a flying-boat serves as a fuselage but has also the additional duties of providing support on the water, together with a suitable medium for alighting and taking off. Its shape must therefore be modified from the true streamline form to cater for these special functions. Nevertheless a well-shaped hull should not show more than twice the drag of a streamline body of similar cross-section or, say, $C_R = 0{\cdot}06$, and by careful shaping it is possible to improve on this. Where the hull is upturned at the stern, to support the tail unit above the water and within the propeller slipstream, the rear portion, or neck, should be of streamlined section in plan.

Seaplane floats are generally given a large fineness ratio for longitudinal stability on the water and this factor prevents the attainment of low resistance, the average drag coefficient value being about double that for hulls.

The inclusion of an undersurface step, for facilitating take-off,* is an adverse resistance feature that is difficult to avoid in both hulls and floats.

The air drag of flying-boat hulls has been investigated† by commencing with an airship shape, then raising the tail to bring the deck horizontal. After this, a Vee bottom was added and finally steps were provided to complete the hull. The percentage drags for the series of models, based on the final shape, were 69, 78, 83 and 100 respectively. The drag increments due to the main and rear steps are roughly 10 and 6 per cent.

The effect of giving the step a Vee shape in plan was tried out and it was found that by sloping back each half-step 30° the drag was increased by 4 per cent., but increasing the angle to 60° brought the drag value back to the original amount. Similarly making the step elliptical in plan produced no drag increase when the rear point was the same as for the large Vee angle, but a 16 per cent. increase resulted from the smaller sweep-back.

The pointed steps gave slightly less water resistance. A step fairing, of length equal to six times the depth, eliminated the step drag altogether, whilst a partial fairing, leaving one-

* See Volume II, Chap. XIII.

† "The Air Drag of Hulls," by L. P. Coombes and K. W. Clark, "Aircraft Engineering," Dec., 1937.

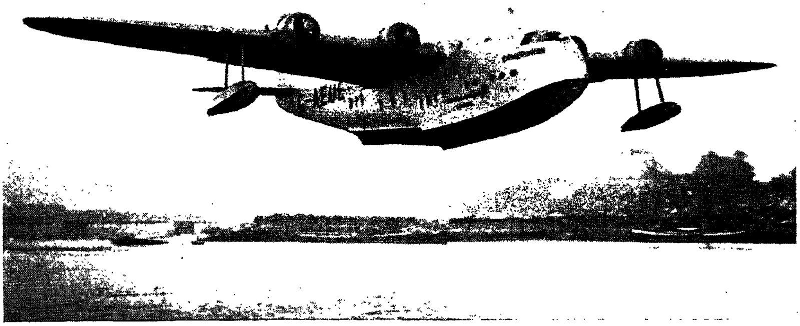


FIG. 131.—SHORT "EMPIRE" FLYING BOAT
FOUR BRISTOL "PEGASUS" XC ENGINES

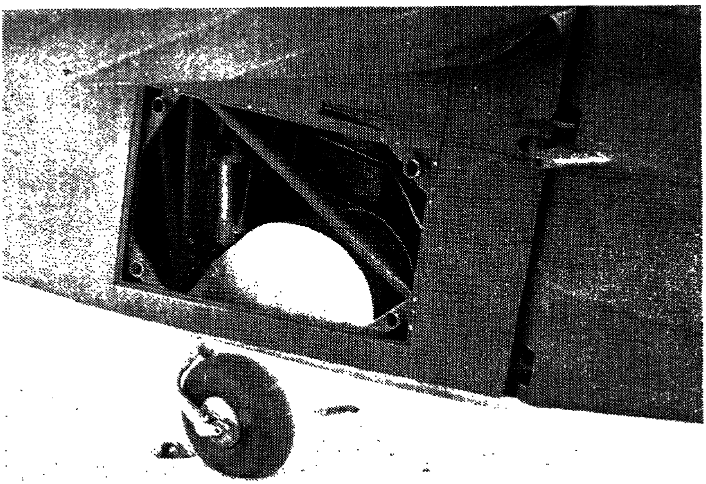


FIG. 132.—RETRACTABLE TAIL WHEEL (INSPECTION COVER REMOVED)
HAWKER "HURRICANE"

(*Reproduced by courtesy of "Flight"*)

quarter of the step exposed, reduced the air drag 10 per cent. without excessive effect on the water performance. A retractable step fairing suggests itself as being a device worth adopting.

Windscreens and Cabin Fronts.

It is difficult to specify the drag effect of an open cockpit since so much depends on the size and shape of the opening, its position on the fuselage and the otherwise cleanness of the fuselage. But the addition of a cockpit opening with the necessary wind-shield may increase the drag of a well-shaped fuselage by as much as 50 per cent. The wind-shield itself has a C_R value of about 0·6, or roughly half the drag coefficient of a flat plate, but may give higher values if the shape is poor and the angle of inclination to the horizontal is excessive. An angle of less than 35° must be avoided if good visibility is to be retained.

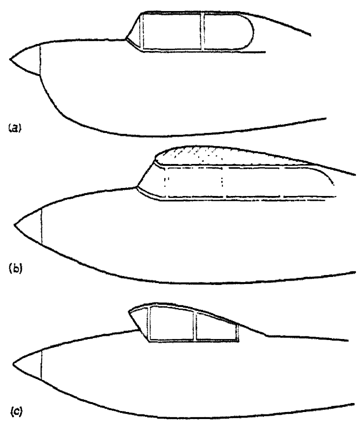


Fig. 133.—Cabin Shapes

The drag coefficient of a cabin front may be taken as approximately 0·4, but since the frontal area is generally wider and deeper than a corresponding open cockpit windscreen the total drag increase may be greater. The undercut shield, as illustrated at (*c*), Fig. 133, gives considerably worse results than the rearward sloping screen and may easily double the drag increment due to the screen.

ENGINE COWLING

Liquid-Cooled Engines.

In the case of liquid-cooled engines the cowling is necessary for fairing purposes only. Generally it should be possible to fair the engine to the fuselage with little resulting increase in resistance.

Radiators.

The liquid-cooled engine requires a radiator for dispersing

the excess heat and this is a drag-forming member of considerable importance. The radiator may be placed in the nose of the fuselage or adjoining the fuselage, generally below, some distance behind the nose, the interference drag being less as the radiator is moved towards the rear.

The drag coefficient of a freely exposed radiator is roughly 0·6, or about one-half that of a flat plate, but there is also the interference effect to be taken into account where the radiator forms part of, or is in close proximity to, the fuselage or other aircraft component, and this may increase the fuselage drag by 100 per cent.

The amount of cooling surface required, i.e., the heat dissipation per unit of area, varies inversely as the speed, whereas the resistance, of course, varies as the square of the speed, so that the importance of drag reduction increases with higher speeds of flight.

By arranging a scoop in front of the radiator more air is forced past the cooling surface, which means that a reduction of radiator area is possible. But the venturi effect, by which the air is speeded up through the honeycomb, results in a nett increase in resistance.

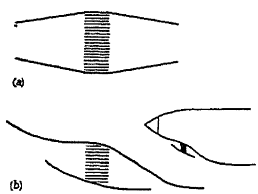


FIG. 134.
RADIATOR COWLING

Conversely the radiator may be housed in a casing which expands from the opening to the radiator position and thereafter contracts again (Fig. 134 (*a*)). The casing retains a smooth flow both within and without, whilst the lowered velocity of the air passing through the honeycomb also contributes towards a reduction of drag.

An increase in the radiator core of roughly 60 per cent. is necessary to compensate for the reduced volume of air passing through, whilst the inlet and exit should have about 60 per cent. of the radiator frontal area for satisfactory results. Despite the increased radiator area the total drag shows a reduction to one-third, with a C_R value of 0·12.

It is possible to obtain similar results for a radiator protruding from a body by arranging the cowling as indicated at (*b*), Fig. 134, and a well-shaped fuselage in which such a radiator has been incorporated should have a C_R of less than 0·1, or, say, one-third that of the fuselage with free radiator.

An expedient for reducing the size of radiator, and consequently drag, is to allow for a certain amount of boiling during the initial climb to the operating height, after which the radiator is of sufficient size for cooling in level flight. This method is referred to as composite cooling and is controlled to some extent by means of adjustable flaps at the exit.

Air-Cooled Engines.

The necessity for the cooling of engines inevitably spells resistance. In the case of an air-cooled cylinder freely exposed to the air-flow, the skin friction drag over the finned cooling surfaces forms but a small part of the total drag set up. The poor aerodynamic shape of the cylinder is accompanied by eddy formation whilst interference and venturi effects also contribute their share to the total drag. Again, as with the radiator just previously considered, the turbulence set up in this case by the exposed cylinders has a very deleterious effect on the drag experienced by the remainder of the fuselage.

For example, the addition of a radial engine to the nose of an otherwise well-streamlined fuselage may multiply its drag by three; or twice in the case of a fuselage initially of poor aerodynamic shape, producing in both cases a drag coefficient of from 0·3 to 0·5. The introduction of crank-case cowling together with a suitable airscrew spinner, one of the early arrangements for combating the high resistance of radial engines, showed a decrease of 10 to 12 per cent. only, a small proportion of the 200 per cent. increase.

The bad drag characteristics of the radial engine at one time almost threatened its extinction, and in fact the special ring cowlings developed in 1928 certainly gave a new lease of life to the air-cooled type of engine.

Cooling and Cowling.

It is now realised that the suppression of drag and the provision of adequate cooling must go hand in hand. Either may be easily achieved by itself but unless both results are attained simultaneously the scheme is unworthy of its place in present-day aeronautics.

The ideal is attained by arranging the cowling in the first instance to complete as nearly as is practicable the streamline

shape of the body to which the engine is attached. Entrance and exit must then be provided to allow *just sufficient* cooling air to enter and leave. This quantity of air should be led with the minimum of disturbances, i.e., obstruction and restriction, direct to the cooling fins and out again to the main stream. Drag may be further decreased by lowering the air speed past the cooling surfaces, which in turn may be achieved by increased fin area and, of course, increased area of duct in this region. It remains to be seen how the desired effect may be carried out.

Helmet Cowling.

This consists of a separate fairing to each cylinder, the air entering through an aperture at the front and escaping through a louvre at the rear (Fig. 135). For adequate cooling there should be internal guides or baffles to lead the air-flow close to the cylinder at the rear as well as at the sides. Helmets have proved very successful for three-cylinder engines and with more attention to the internal details it is even probable that they may supersede the ring cowling for seven- and nine-cylinder engines. It is possible to obtain improved conditions by arranging for the exit to be situated at the forward curved part of the cowl, and so take advantage of the lowered pressure at that point.

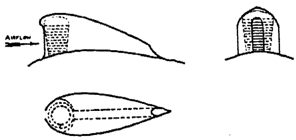


FIG. 135.—HELMET COWLING

Cylinder-in-Line Cowling.

For the cylinder-in-line engine air is introduced through a vent facing forward and is led by a duct to one side of the line of cylinders, passes between the cylinders and escapes through openings at the opposite side (Fig. 136). Baffles are inserted between the cylinders on the egress side, to ensure complete circulation, whilst the air is prevented from escaping upwards or downwards by horizontal baffle plates situated at the base and head of the cylinders.

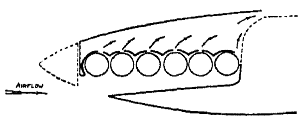


FIG. 136.—COWLING OF CYLINDER IN-LINE ENGINE

Outer cowling follows the lines of the fuselage, or nacelle,

and between this and the cooling duct cowling may be housed the engine ancillaries such as carburettors and magnetos.

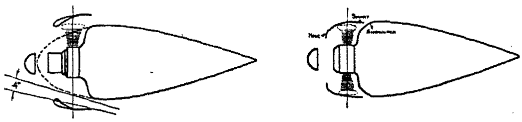
Radial Engine Ring Cowling.

The ring cowling serves first as a means of preventing the break-away of air-flow due to the bluff nose caused by a radial engine, and is accomplished by the presence of an annular cowl of more or less aerofoil section. The cooling is obtained by the introduction of a relatively small amount of air which is fed at a greatly reduced velocity to those parts of the engine where the heat present is excessive.

The ring cowling may be divided into two types, the Townend ring and the N.A.C.A. cowl, the former having been developed in this country simultaneously with the development of the latter in America.

Townend Ring.

The Townend ring (Fig. 137 (*a*)) consists of an aerofoil section ring on which is produced a radially outward and forward force with accompanying downwash of the air-flow. The ring should be of high-lift section with about 10 per cent. or greater camber, maximum camber being at the mid-chord location, and of single surface, except for the leading-edge portion where it has been found that beneficial results may be obtained with a bulbous nose. This nose has been put to practical use as an exhaust gas collector.



(a) (b)

FIG. 137.—TOWNEND RING AND N.A.C.A. COWLING

The angular setting of the ring chord may be obtained as follows : The streamlines over the nose of the body with the engine removed and the nose portion properly faired may be calculated or ascertained by means of a smoke tunnel, or streamers. The ring is then set with the no-lift line parallel to a tangent to the streamline at the mid-ring position. Alternatively, the ring chord is set at an angle of incidence of 4° to the tangent to the

body (faired forward) at the mid-ring position (see Fig. 137 (*a*)). These settings give minimum drag and avoid stalling of the ring over the normal flight range.

The ring is placed so as to extend equally in front of and behind the engine centre line, or just forward of this position. Slightly better results have been obtained with a polygonal ring in place of the circular ring.

The fuselage diameter immediately to the rear of the ring should be about 0·75 of the engine diameter, whilst the ring chord should be 0·5 of the engine diameter. The body just rearward of the ring should be circular, or polygonal if a polygonal ring is fitted, with but small convergence over a length roughly equal to the chord of the ring. Otherwise the body shape has little effect on the ring characteristics.

The aerodynamic effect of the Townend ring* is to reduce the fuselage C_R to 0·14 though this is accomplished only at the expense of some reduction, not necessarily serious, of the cooling properties.

The Townend ring has also been utilised in conjunction with pusher engines, the beneficial drag results obtained to date being approximately one-half those for the tractor arrangement. In this instance the cooling is improved.

N.A.C.A. Cowl.

This cowling is shown in Fig. 137 (*b*). Its object is to complete the fairing as far as possible from the fuselage shoulder to the airscrew boss. Annular spaces are left at the front and rear of the cowl sufficient only to allow the required minimum volume of cooling air to enter and leave.

The cowl contour is approximately elliptical, with the leading-edge tangential to a plane parallel to the plane of the engine, and with the skirt aligned with the body surface to the rear of the cowl. The diameter of the entrance should be from 65 to 75 per cent. of the engine diameter for single-bank engines and from 75 to 85 per cent. for double-bank engines, or alternatively the entrance area should be about 3 sq. in. per horse-power for an aircraft with a climbing speed of 100 m.p.h., increasing for slower climbing speeds. When baffles are fitted (see page 199) the entry may be reduced by one-quarter with decrease of drag and no sacrifice of engine cooling.

* " Engine Cowling," J. D. North, *R.Ae.S. Journal*, July, 1934.

The exit area should be from 0·4 to 0·5 that of the inlet (the smaller figure relating to cowls fitted with baffles), the decreased area causing a speeding up of the departing air. Enlargement of the exit annulus gives considerable increase of the volume of air flowing through with small increase of drag.

Best results have been obtained by filling the spaces between the cylinders with baffles (Fig. 138), and so avoiding the passage of non-cooling air. In other words the whole of the air passing through is fully utilised and thus the total volume may be reduced with consequent reduction of drag.

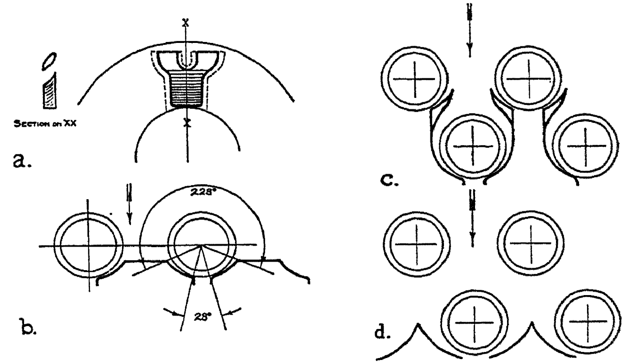


FIG. 138.—BAFFLES FOR N.A.C.A. COWL

Baffle arrangements and dimensions are shown at (*a*) and (*b*), Fig. 138, for single-bank engines, and at (*c*) and (*d*) for double-banks. The clearance between the baffle and cylinder fins should be about $\frac{1}{4}$ in. to $\frac{3}{8}$ in. at the foremost point, decreasing to $\frac{1}{8}$ in. or less at the rear.

The increased drag of a fuselage caused by the presence of a radial engine fitted with the N.A.C.A. cowl and baffles is 70 to 73 per cent., or less than one-third of the increase due to the engine alone. The best fuselage C_R value so far attained is 0·155 and in the case of a nacelle, of fineness ratio 2·3, a C_R of but 0·135 has been reached.

Measured in terms of the increase of speed* an aeroplane fitted with the cowl showed an increase of 8·5 per cent. The addition of baffles gave a further 0·5 per cent. with improved cooling. Re-designed for equal cooling, the new cowling with

* "The Cowling and Cooling of Radial Air-cooled Aircraft Engines," Beisel, MacClain and Thomas, *R.Ae.S. Journal*, July, 1934.

baffles gave a further 4·0 per cent. increase, or a total increase of 13 per cent.

Still better results may be attained by the provision of ducts for leading the air to the cylinders and away after passing the baffles. Further improvement, as already mentioned, may be obtained by arranging the exit at the front, curved part of the cowl in order to benefit from the lowered pressure there. Cooling is not adversely affected by the N.A.C.A. cowl, but in certain cases has been definitely improved.

Temperature control to allow for the varying conditions of flight has been obtained by arranging hinged flaps to the skirt of the cowl, controllable from the cockpit. Thus during a climb, when the flight speed is relatively low and the engine-power required is large, the flaps may be opened to increase the volume of air passing through the cowl at the expense of a small loss of speed.

In general the N.A.C.A. cowl appears to be superior to the Townend ring, though there is little to choose between the two as regards drag. For fuselages of small cross-section the Townend ring might prove more suitable but where the cross-section is large, as with cabin machines, the N.A.C.A. cowl cannot be surpassed. The same applies to engine nacelles housed in wings.

Expressed in terms of the engine-power, the cooling drag accounts for roughly 6 per cent. of the brake horse-power in present-day designs.

Circular Struts, Wires and Cables.

The variation of C_R of cylinders according to Reynolds' number is shown by the curves of Fig. 139. The curve is irregular, as was explained earlier, and shows a rapid decrease of drag over the transition period, marked T in the diagram. The full curve represents smooth flow conditions of the free air-flow for which the transition occurs between the limits of 10^5 and 5×10^5. Where turbulence is present the critical region commences at a lower value of R.N., roughly 50,000 for maximum turbulence, as shown by the dotted

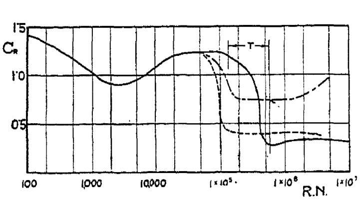


Fig. 139.—Drag of Cylinders at Varying Reynolds' Numbers

curve. Surface roughness has a similar effect, depending on the degree of roughness, but does not produce so large a decrease of C_R, whilst beyond a certain R.N. value the drag commences to rise again. This has been indicated by the chain curve. The minimum value of C_R, about 0·3, is seen to be at high R.N., but there are probably few instances in practice where this value is attained.

Circular tubes and rods are seldom used in exposed places on aircraft, more particularly at high Reynolds' numbers. Unless therefore R.N. exceeds 5×10^5, C_R may be taken approximately as 1·1. In the case of wires of circular section and cables, still lower values of R.N. are likely to obtain, and again C_R may be taken as 1·1. At the lower end of the R.N. scale for circular wires, say, at 3,000, represented by a 16 or 14 S.W.G. wire at a speed of 60 or 70 m.p.h., a lower drag coefficient of 0·9 should be used.

Stranded cable has been found to have similar drag characteristics to round wire, but at high R.N., 5×10^5 and over, the uneven surface results in decreased C_R, but at still higher values of R.N., certainly above 5×10^5, the drag of cable is likely to be greater than that of smooth wire.

Struts of Streamline Section.

Until more information becomes available the aerofoil R.A.F.30, modified to give a fineness ratio of about 3·0 or 3·5, and with rounding off of the trailing-edge, may be taken as being little, if any, inferior to any other strut section.

This section has moreover a relatively high moment of inertia about the major axis, a feature of importance in the structural design of struts.

Since the resistance of streamline struts has a large skin friction component, the Drag/R.N. curve has a likeness to the skin friction curve of Fig. 130 (page 188). C_R ranges from about 0·1 at a Reynolds' number of 10^5, falls to a minimum of 0·06 at from 5×10^5 to 10^6, and rises thence to 0·08 at 2×10^6. Some reduction takes place with further increase of R.N., but this is beyond the normal flight conditions for struts.

Circular tubes are often used for struts, but with added fairing to give a streamline section. At (*a*), Fig. 140, is shown a simple but inefficient fairing, which, however, may give a drag reduction of from one-half to three-quarters; (*b*) is an improved type of tail fairing, which despite the increased width of section should

decrease drag to one-fifth that of the original strut. The best solution, aerodynamically, is obtained by adding a nose as well as the tail fairing, as at (*c*), but this adds also to the total of manufacturing expense. Built-up struts should be given a fineness ratio of at least 3·0.

Streamline Wires.

The most usual form is the lenticular, symmetrical about both axes, with a fineness ratio of 4, giving a C_R of 0·35. Wires of true streamline section are sometimes used on racing aircraft, for which the drag is halved at high Reynolds' numbers (for wires, 10^5), but little advantage is gained at low values of R.N.

Duplicated Struts and Wires.

When circular struts or wires are placed one behind the other in the line of flight, the rear member is shielded to some extent

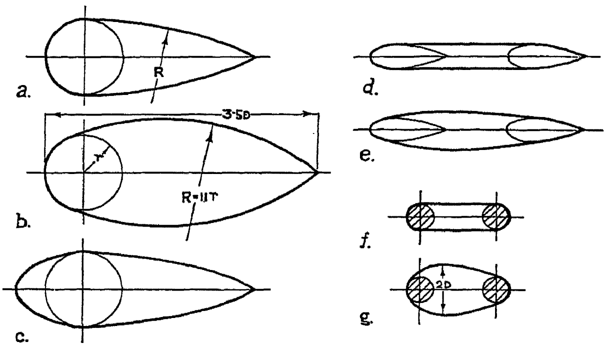


FIG. 140.—FAIRINGS OF STRUTS AND WIRES

by the other, with a consequent reduction of total drag. Where two wires are close together the drag is roughly half that for a single wire, and with two diameters separation the drag is still equal only to single wire drag. Even up to 10 diameters there may be some saving, but above 6 the total drag decrease is slight. With lenticular wires the drag is halved when they are in close proximity, but the saving is quickly lost with increased gap and is of little importance when the separation exceeds the sectional length of the wire.

Streamline struts in tandem, on the other hand, create an

adverse drag effect, and up to a separation of about eight times the section width it is advantageous to enclose the two within a fairing ((*d*) and (*e*) of Fig. 140), the latter of course being superior.

As might be anticipated still better results are obtainable by enclosing circular tandem struts or wires within a fairing.

The simple fairing (*f*) of Fig. 140 gives a total drag of one-half single-wire drag up to a separation of 7 diameters, and even better is the streamlined fairing shown at (*g*), which gives an overall drag of little more than one-quarter that for a single exposed wire.

Inclination of Struts and Wires.

The effect of placing struts and wires of circular section with their longitudinal axis at some angle to the normal to the direction of flight, or relative air-flow, is to decrease the drag per unit length by roughly 4 per cent. for a 10° inclination, rising to 15 per cent. at 20°, 30 per cent. at 30°, and 50 per cent. at 40°, due both to the reduction of frontal projected area and elongation of the effective section. The effect is not so pronounced in the case of streamline struts, as is only to be expected, the reduction of drag being about three-fifths that for round struts.

The interference drag of struts is dealt with in a later section.

TABLE VII.—DRAG OF AIRCRAFT COMPONENTS

Item	Drag in lb. per sq. ft. at 100 m.p.h.
Flat plate	30·0
Streamline form	0·7–1·5
Fuselage	2·0–4·0
Flying-boat hull	2·5–4·0
Seaplane floats	4·0–6·0
Circular wires, cables, and struts	8·0–27·0
Lenticular wires	1·5–2·5
Streamline struts	8·0–10·0

Control Surface Breaks.

The gap caused by the hinging of control surfaces, ailerons, elevators, and rudders, gives rise to an increased drag amounting to from 5 to 15 per cent. of the minimum drag of that part of the

span affected. The lower figure relates to well-shrouded gaps and the higher figure corresponds to gaps with no shrouding. It should be noted that control power is also adversely affected by gaps through which the air may flow unrestricted.

Table VII has been prepared to facilitate rapid calculations of drag, but since so much depends on the R.N. relative to the actual conditions the figures given should be used with caution. One point brought out by the table is the high relative drag of all exposed wires and cables, and the suppression of such members is highly desirable.

Landing Wheels and Undercarriages.

High-speed aircraft are generally fitted with retractable landing wheels, which diminishes to some extent the importance of this considerable drag item. For relatively slow aeroplanes the complication and weight of retracting gear is seldom considered worth the saving in resistance, especially if proper attention is given to ensure efficient fairing of the undercarriage details. It may also be remembered that retractable gear is extended during the take-off and initial climb and therefore such aircraft suffer under a disadvantage compared with those having well-faired fixed landing gear.

In the earlier retracting gears, drag in the lowered position was not considered important, in fact an untidy appearance was often purposely adopted for steepening the glide in the landing approach. It is now realised that for a good take-off the extended gear should be aerodynamically clean, landing flaps being relied upon for providing the required landing qualities.

Systematic research on various types of landing chassis has been carried out in America,* from which most of the following information has been extracted.

There is little to choose as regards drag between unfaired high pressure, medium pressure, and balloon wheels of similar weight-carrying capacity, and the shorter structure required for the first-mentioned probably gives this type the advantage. Streamline section wheels, however, the latest arrival to the family, show a reduction of drag amounting to roughly 20 per cent. On the other hand, wired high pressure wheels, without side discs, show a drag increase of over 60 per cent.

* "The Drag of Airplane Wheels, Wheel Fairings, and Landing Gears," *N.A.C.A.* Tech. Reports 485, 518 and 522.

The drag of wheels is generally stated in lb. at 100 m.p.h., the appropriate figure for the wheel shown in Fig. 141 being 9·5 lb. A suitable fairing, as shown at (*a*), reduces the drag by some 42 per cent. to 5·5 lb.

Extending the research to practical conditions, lengths of streamline struts were attached to the wheel to represent "vee" and "split" axle type undercarriages, the free strut drag amounting to 3·2 lb. per wheel. Drag measurement of the combination wheel and struts amounted, however, to 18 lb., and with the wheel fairing 15·9, showing increases due to interference of 5·3 and 7·2 lb. respectively, or 42 and 83 per cent.

By paying careful attention to details, e.g., fairing off all strut junctions, interference effects may be considerably reduced. As a fairly approximate guide the free drag of the faired wheels and struts may be added, and, assuming all joints and fittings are enclosed, the combined drag should be increased by 5 per cent. for each node in the landing gear structure.

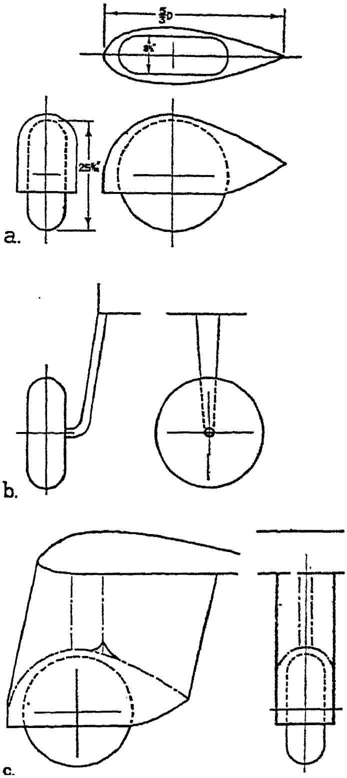


Fig. 141.—Landing Wheel Fairings

The simplified landing gear illustrated at (*b*) has a total drag of 23·5 lb. With streamline wheels substituted for the medium pressure wheels used in the test the drag would probably be under 20 lb. The addition of wheel fairing reduced the figure to 17·5 lb., of which interference accounted for but 14 per cent., or an increase of 16·3 per cent. over the nett drag, compared with 20·0 per cent. as might be computed according to the method given above.

The type of landing gear fitted to low-wing monoplanes is shown at (*c*), Fig. 141. With either of the fairings, outlined

in full and broken lines respectively, the drag measured but 13 lb., there being apparently nothing to choose between these two excellent designs, but it is important to note that the removal of the expanding fillet between the strut and wheel fairing added a further 7 lb. of drag, an alarming penalty for what would appear to be a very small divergence from aerodynamic cleanliness. This also suggests that a moderate angle of pitch might still result in a disproportional addition of drag for this lay-out so that the "trousered" arrangement may prove superior in actual use.

The table below is added for rapidity of reference, and relates to a basic wheel of $8\frac{1}{2}$ in. $\times$ $25\frac{5}{16}$ in. and an air speed of 100 m.p.h.

TABLE VIII.—DRAG OF VARIOUS UNDERCARRIAGE ARRANGEMENTS

Component	Drag in lb. at 100 m.p.h.
Wheel alone	9·5
Wheel with fairings, (*a*) Fig. 141	5·5
Undercarriage struts alone	6·4
2 wheels and struts	36·0
2 wheels, fairings and struts	31·8
2 wheels and struts, (*b*) Fig. 141	23·5
2 wheels faired and struts, (*b*) Fig. 141	17·5
2 wheels and fairings, (*c*) Fig. 141, either . .	13·0

Interference Drag.

Some reference has already been made to the effect on drag of two bodies, or components, being in close proximity to each other. In the case of struts and wires in tandem, the effect has been seen to be beneficial. In the case of undercarriage components the reverse holds good. Generally, the results of interference are adverse, but by careful attention to detail and relative positioning of parts adjoining, or in close proximity to, other parts, it is possible to minimise such drag losses and in many cases to turn the effects into a gain.

The drag increase, due to some excrescence, of a body approximating to true streamline shape is least when the fineness ratio

is high. Interference drag is a maximum when the body causing interference is situated at, or near the largest diameter, or section.

Fuselage–Wing Interference.

One of the most important aspects of interference is that due to the placing together of the main plane and fuselage, the interference affecting both lift and drag, and sometimes stability and control as well. One of the earliest comprehensive series of tests was carried out in Canada in 1927* in which no less than 65 wing–fuselage combinations were tested in the wind tunnel. A further series of tests was made in this country† and more recently a further 209 combinations have been tested in America.‡

In general, the drag is lowest for the mid-wing position, it being then about 5 per cent. less than the combined individual drags of wing and fuselage, or roughly equal to the drags of the fuselage and the *exposed* part of the wing. Raising or lowering the wing results, at first, in slight drag increase but as the top and bottom positions are reached the increase becomes rapid and continues until the wing chord is approximately one-eighth of the fuselage depth clear of the upper and lower fuselage boundaries. Further separation of the two components causes a rapid decrease in drag, and when there is a clearance of one-half depth of the fuselage, the total drag may be slightly less than the combined free drags, particularly in the parasol arrangement.

With the normal high-wing or low-wing monoplane, in which the wing surfaces do not protrude beyond the fuselage contour, the drag may be taken as equal to the combined drags of the fuselage and *full* area wing. Just external to the fuselage, that is resting on top of the fuselage or attached immediately below, the drag is increased by some 25 per cent. at low incidence, increasing still further with increased incidence at the low position but decreasing somewhat for the high-wing setting. These figures relate chiefly to fuselages of round or oval cross-section. With a square-sided, or rectangular, fuselage the adverse results are not so pronounced and may be taken as approximately half those as detailed above. No account has so far been taken

* "The Interference between the Body and Wings of Aircraft," by J. H. Parkin and G. J. Klein, *R.Ae.S. Journal*, Jan., 1930.

† *R. & M.* No. 1480. Also *R.Ae.S. Journal*, July, 1932.

‡ "Interference of Wing and Fuselage from Tests of 209 Combinations in the N.A.C.A. Variable-Density Tunnel," *N.A.C.A.* Report No. 540.

of the effect of inserting fillets between the wing-root and fuselage, as this is dealt with in a later paragraph.

Interference effects on lift are greatest for the low-wing where the loss of maximum lift is about 10 per cent., decreasing to about 5 per cent. for the mid-wing arrangement, still less for the high-wing and zero for the parasol monoplane. These figures are based on the *full* wing area, which means that the interference loss of lift on the parasol wing is roughly compensated for by the lift contributed by the fuselage.

It has been suggested* that the interference effects considered, are caused by the alteration of effective wing camber due to the curved air-flow around the fuselage with consequent decreased camber as the wing is raised relative to the body. This, however, does not agree with all the changes in lift and drag, and moreover, tests with an aerofoil of symmetrical section, having no mean camber, showed greatest interference effects. The true explanation may be found in the upsetting of the air-flow with consequent burbling over the vital upper surface of the wing, this being, of course, more pronounced for low-wing settings.

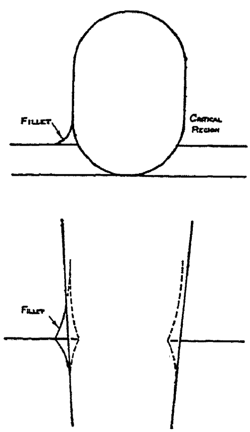


Fig. 142
Wing Root Fillet

Closer examination reveals a critical region on the upper side of the wing-root, bounded by the fuselage side and the upper wing surface, and it is within this region that most of the ill effects have their origin, the cause being two-fold (Fig. 142). First, the air-flow here has greater frictional resistance due to the double bounding walls, and is therefore slowed down. The amount of bounding surface, with consequent loss of velocity, are maxima in the combination formed by a low wing and circular section fuselage, or any other fuselage in which the width decreases with depth above the level of the wing. The second point concerns the fuselage width at the wing-root, where it generally commences to taper towards the tail, and it will be readily appreciated that this has the effect of increasing the air-flow area with loss of velocity, the effect again being aggravated with a low wing and circular fuselage

* Ref. p. 207 (1).

on account of the downward curvature of the rear upper surface of the wing.

The two factors outlined both produce pressure increase and burbling,* and it is only by careful attention to the shaping at this junction of the two surfaces that the detrimental characteristics may be overcome in a reasonably satisfactory manner.

The obvious solution lies in the provision of some sort of a wing-root fillet and in many examples the fillet takes the form of a quarter-circle of gradually increasing radius from the leading-edge to 0·25 chord at the trailing-edge for a low wing/circular fuselage combination, but of lesser amount for more favourable junctions. The shape of the fillet should be so arranged to prevent any increase in area above the wing, due either to cross-sectional curvature of the fuselage or tapering towards the rear. Behind the trailing-edge, the fillet should be faired off to prevent upsetting of the flow (Fig. 142).

Cases are known in which the removal of root fillets has resulted in improved stability and even performance, and if expansion at the wing junction can be prevented without resort to fillets, the best results are likely to be obtained.

Engine Nacelle–Wing Interference.

The interference between a nacelle and wing is not limited to lift and drag, but is further complicated by the presence of the airscrew, which also introduces mutual interference between the wing and airscrew. A great deal of research has been undertaken of late in America† on the important question of optimum nacelle location, from which the following brief results have been extracted.

Considering first the drag of a nacelle when placed in various positions above, below, and in the wing, the last mentioned position proves greatly superior to the others, having a drag of from 20 to 40 per cent. less than the free nacelle drag, according to the longtudinal setting relative to the wing. The worst combination is that of an engine nacelle above the wing, due of course to the upsetting of the lift grading curve with consequent increase of induced drag.

With the airscrew running, there is an increased velocity of flow over the wing which augments the lift, or in other words,

* See p. 29 (Chap. III).
† *N.A.C.A.* Tech. Note 320. See also *R. & M.* No. 1414.

enables the wing incidence to be slightly decreased to give an unaltered lift value.

For maximum propulsive efficiency, the nacelle should be placed below the wing and well forward, but it will now be realised that the optimum position must be a compromise between the settings for lowest drag and greatest propulsion. In other words, the desired position is the one that shows the highest nett thrust after the deduction of nacelle drag, and here again, the nacelle sunk into the wing leading-edge is markedly superior, particularly at the higher speeds of flight. In general, the best longitudinal location is obtained by setting the airscrew axis about one-third of the wing chord forward of the leading-edge. At the slower speeds, as for instance during the climb, a more forward position becomes rather better and at still lower speeds the nacelle slung under the wing shows up to advantage.

The high engine position has been shown to be so inferior, and the low position also for high speed, that they should be rigorously avoided, at least until further research may have revealed ways of overcoming the poor aerodynamic qualities exhibited by external nacelles.

Strut Junction Interference.

The interference drag of a strut mounted on the upper surface of a wing is equal to the drag of a considerable length of strut, the interference increasing rapidly as the strut angle with the wing span is decreased. As would be expected, the interference due to struts attached to the lower surface of a wing is considerably less.

This strut attachment interference, as also the presence of any other body or part extending beyond the upper wing surface, is a cause of premature stalling.

THE IDEAL STREAMLINE AEROPLANE

The conception of the ideal streamline aeroplane was first put forward by Melvill Jones in 1929* as one in which the drag consists of the sum of the induced drag and the drag due to skin friction over the body and other essential parts.

Aeroplanes of poor aerodynamic shape, and whose parts

* "The Streamline Aeroplane," by Professor B. Melvill Jones, *R.Ae.S. Journal*, May, 1929.

FIG. 143.—VICKERS-SUPERMARINE "SPITFIRE"
ROLLS-ROYCE "MERLIN" ENGINE
(Reproduced by courtesy of "Flight")

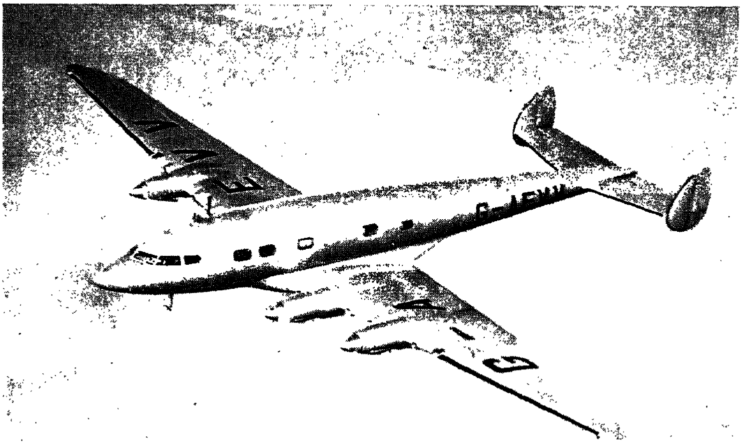


FIG. 144. DE HAVILLAND ALBATROSS AIR LINER
FOUR "GIPSY TWELVE" ENGINES
(Reproduced by courtesy of "The Aeroplane")

are not carefully faired one to another, prevent the attainment of streamline flow and give rise to the shedding of continuous streams of eddies during flight. This eddy formation represents wasted energy and requires power expenditure in excess of that required to overcome the unavoidable drag mentioned above, often to the extent of three or more times. Such excess is a measure of the inefficiency of the aerodynamic design.

The induced drag is the price paid for lift and cannot be avoided. Only by increasing the span, rendered difficult by structural considerations, may induced drag be decreased. Since, however, at the present time at least, induced drag represents a small part of the total head resistance, its further suppression need not be considered of great importance. It remains, then, to concentrate on the proper streamlining of the remainder of the aircraft.

A true streamline body is one around which the flow of air approximates closely to that of the hypothetical inviscid fluid, except within the thin boundary layer. Under such conditions the only drag present is due to the tangential surface forces, called skin friction. The total drag of the ideally streamlined aeroplane may then be expressed as

$$D = D_i + D_f$$

where D_i = induced drag and
D_f = skin friction drag.

Calculation of the induced drag component for any one aeroplane is simply made from the formula

$$D_i = \frac{L^2}{\frac{\rho}{2} \pi b^2 V^2} \quad . \quad . \quad . \quad . \quad . \quad . \quad . \quad (44)$$

To obtain the skin friction drag, Melvill Jones has shown that it is sufficient, within reasonable limits, to compute the "wetted," or exposed, surface of the aircraft and to treat it as if it were a flat plate subjected to flow parallel to its face. The drag coefficient to be employed must depend on the appropriate Reynolds' number, and for present-day fuselages and wings, but excluding any small items, such as lift struts for which there is no place in the ideal aeroplane, R.N. varies between 4×10^6 and 5×10^7. It will be seen from Fig. 130 that the boundary

layer flow is likely to be turbulent and the values for the R.N. stated are 0·004 and 0·003. It is suggested that the higher value should be adopted as an overall figure for C_{Df}. The value of E, the equivalent flat plate area, has been found to lie between 3·0 and 3·5 S with 3·2 S as a fair average value.

Adopting these figures the allowable skin friction drag of the ideal streamline aeroplane is given by

$$D_f = 0{\cdot}0128 \frac{\rho}{2} SV^2 \tag{45}$$

If greater accuracy should be desired, allowance may be made for those parts of the aircraft subjected to the higher velocity within the slipstream, but it should be noted that the value assumed for C_{Df} was arbitrarily chosen and is insufficiently accurate for such refinement.

The ideal streamline aeroplane does not exist, most present-day aircraft producing two to three times the ideal resistance, which means that from half to two-thirds of the power is expended in the generation of unnecessary turbulence, though there has been a marked cleaning-up of design during the past few years. Like all ideals, the truly streamline aeroplane may be difficult of attainment, but at least it provides a standard of streamlining to be striven for and one by which the performance of aircraft may be measured.

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